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ORBITAL SPACECRAFT CONSUMABLES RESUPPLY SYSTEM (OSCRS)

FINAL REPORT
VOLUME IV
EXTENDED STUDY RESULTS

(DRD-10)

Prepared for
the
National Aeronautics and Space Administration
Lyndon B. Johnson Space Center

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FOREWORD

This final report of the Orbital Spacecraft Consumables Resupply System (OSCRS) study was prepared by the Space Transportation Systems Division of Rockwell International for the National Aeronautics and Space Administration, Johnson Space Center, Houston, Texas, in compliance with the requirements of Contract NAS9-17584, CDRL No. MA 1023T.

In response with the CDRL instructions, this report was submitted in three separately bound volumes. A fourth volume is added for the extended Study.

- Vol. 1. Executive Summary
- Vol. 2. Study Results
- Vol. 3. Program Cost Estimate
- Vol. 4. Extended Study Results

Further information concerning the contents of this report may be obtained from D. Perry, Study Program Manager, telephone (213) 922-2584, Downey, California.

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PART I
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SEPTEMBER 1987

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Space Transportation
Systems Division

EXECUTIVE SUMMARY

1.0 Executive Summary Introduction

This report summarizes the results of the Orbital Spacecraft Consumable Resupply System (OSCRS) study performed by Rockwell International for the National Aeronautics and Space Administration (NASA) at Johnson Space Center (JSC) under contract NAS9-17584. The study was performed in accordance with the study plan contained in modification 5C to the contract NAS9-17584. The study plan was set up to follow the Contract Change Authorization (CCA) SOW which expanded the basic contract SOW subtasks 1.1, 1.2, 1.3, 2.1.1, 2.1.2, 2.1.3, 2.2, 2.3 and 4.1 as shown in the schedule depicted in Figure 1.0-1.

The objectives of this extended study consist of three major tasks. The first task is to establish the definition of Space Station and OMV user requirements and interfaces and to evaluate system requirements of a water tanker to be used at the Station. The second task is to conduct trade studies of system requirements, hardware/software and operations to evaluate the effect of automatic operation at the Station or remote from the Station in consonance with the OMV. The results of the trade studies are used to; 1) establish the Station/OMV/OSCRS interfaces in an interface control document (ICD); 2) establish a revised phase C/D program schedule; and 3) estimate phase C/D costs to incorporate unique Station/OMV features into the OSCRS. The last task is to evaluate automatic refueling concepts for use in the Orbiter, at the Space Station or remotely with the OMV and to evaluate the impact to OSCRS concept/design to use expendable launch vehicles (ELV's) to place the tanker into orbit.

2.0 Executive Summary Study Conclusions

The OSCRS extended study consisted of three statement of work tasks. These tasks were performed in accordance with the extended study plan to the schedule shown in Figure 1.0-1. The second portion of this volume contains the detailed results of these trade studies. The following discussion summarizes the results and conclusions reached in each of the following study areas:

- Users Requirements
- Water Subsystem
- Automated versus EVA
- Storage Capability
- Offloading Residuals
- Thermal Effects
- On-Orbit Operations
- Shuttle to Station Transfer
- Central versus Multilocations
- OSCRS/OMV Usage
- OSCRS Launch via ELV
- Automatic Interface Design

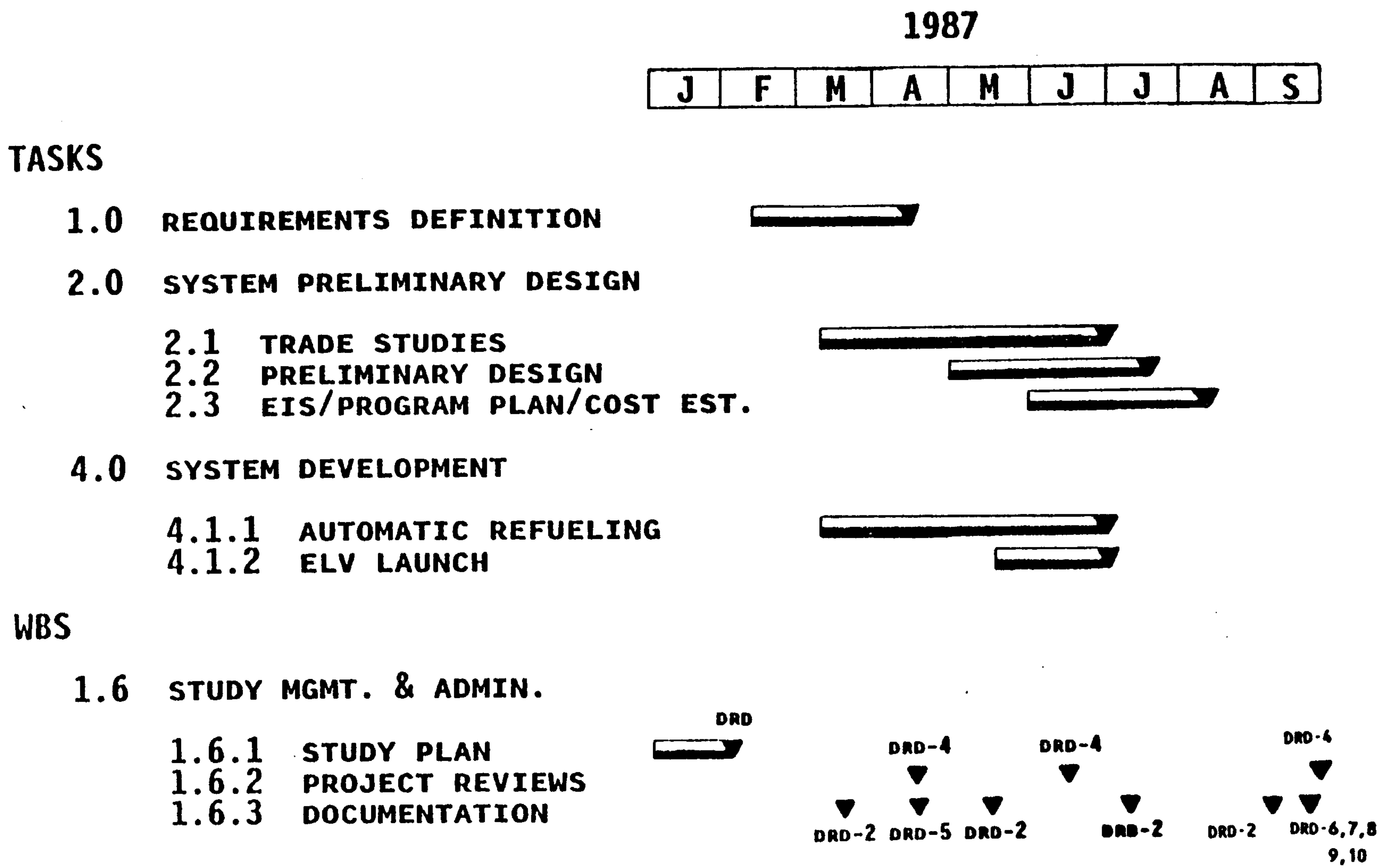


Figure 1.0-1 - OSCRS Study Extension Task Schedule

2.1 User Requirements Definition

Data from existing Rockwell data bases, NASA-JSC, and business contacts with potential resupply candidates were used to determine resupply quantities and schedules per mission for storable propellants (Table 2.1-2). Potential space station resupply missions are indicated by an asterisk. The space station fluids to be resupplied include hydrazine, bipropellants, water, and gaseous nitrogen and helium. Figure 2.1-1 details the quantities of the different propellant types to be resupplied at the space station on a yearly basis. Table 2.1-1 lists the yearly averaged fluid resupply quantities to be resupplied to or from the space station, including storable propellant, water, gaseous nitrogen and helium.

TABLE 2.1-1. SPACE STATION YEARLY AVERAGED RESUPPLY QUANTITIES

FLUID	QUANTITY (LBM)	RESUPPLIED TO
Hydrazine (N2H4)	3500	Spacecraft, OMV
Bipropellant	3500	Spacecraft
Water	6040	Space Station
GN2	840	OMV, Spacecraft
GHe	100	Spacecraft

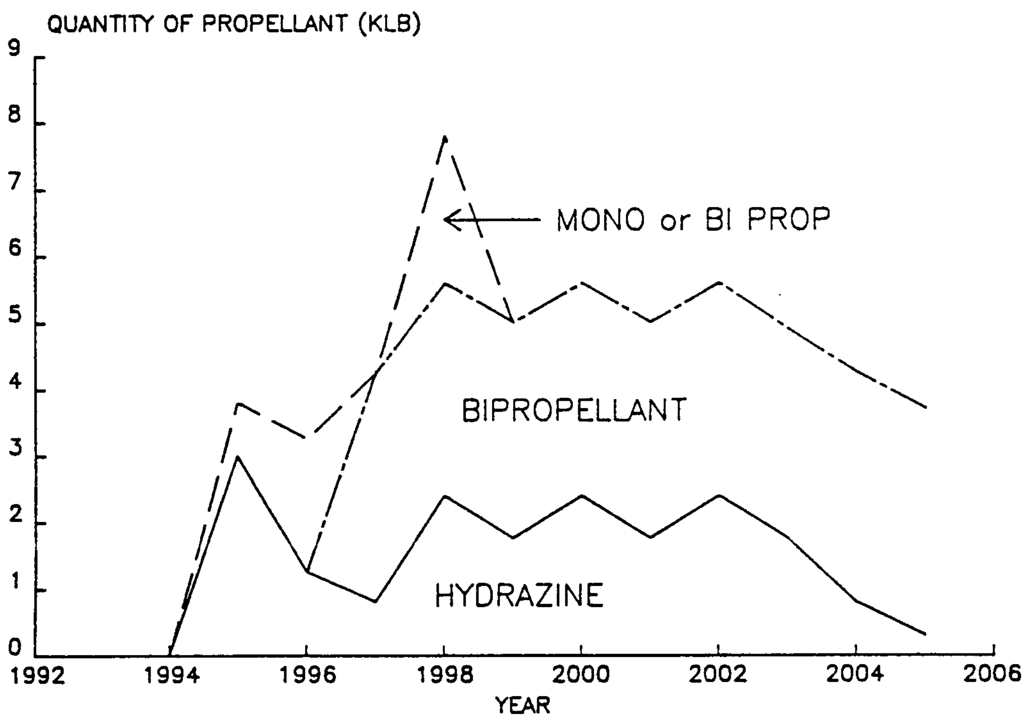
2.2 Preliminary Water Tanker System Requirements

The preliminary monopropellant OSCRS tanker was designed as a generic tanker. Using the tanker to resupply water does not present any unexpected impacts to design. The primary structure will remain unchanged with minor modifications to the secondary structure because of the smaller water subsystem. The avionics subsystem was designed for a more complicated propellant transfer, thus it is ample for a simpler water transfer. Monitoring a nonexplosive fluid reduces the need for extensive quantities of temperature sensors. A reduction from 102 to 55 temperature sensors occurs when the water subsystem replaces the hydrazine subsystem. The water subsystem is designed to carry 2300 lbs of water (with two water tanks as shown in Figure 2.2-1). A comparison to the hydrazine subsystem shows a 161 lb decrease or a dry weight of 293 lb for the water subsystem.

Table 2.1-2 - Storable Propellant Resupply Requirements

MISSION NAME	POTENTIAL SS RESUPPLY	ALT (NMI)	INC. (DEG)	LAUNCH DATE	RESUPPLY SCHEDULE (DAYS)	QUANTITY
GRAVITATIONAL WAVE DETECTOR	•	220	28.5	2001	180 (FOR 2 YR)	66 LBS N2H4
GRO	•	200	28.5	1991	730	2480 LBS N2H4
M-SAT-B		19310	0	1994	(1994)	1100 LBS N2H4
M-SAT-C		19310	0	2000	(2002)	2200 LBS N2H4
EXPLORER PLATFORM	•	220	28.5	1995	365 (FOR 9 YR)	551 LBS N2H4
TOPEX		720	64	1996	(1999 & 2002)	300 LBS N2H4
EARTH OBSERVATION SYSTEM		445	98.7	1996	365 (FOR 7 YR)	1800 LBS N2H4
EARTH RESOURCES SATELLITE		381	98.7	1995,98,01	(1997,00,03)	200 LBS N2H4
ASTRONOMICAL PLATFORM	•	220	28.5	1997	180 (FOR 7 YR)	397 LBS N2H4
SPOT		450	98.7	1993	1460	500 LBS N2H4
DOD D		450	98.7	1991,92,93,94	1825 (FOR 10 YR)	70 LBS N2H4
NAVY REMOTE SENSING SYSTEM		450	98.7	1990	1095 (FOR 12 YR)	70 LBS N2H4
EURECA	•	270	28.5	1994	730 (FOR 8 YR)	700 LBS N2H4
DOD A		125	96.5	1992 (2)	730 (FOR 10 YR)	7000 LBS NTO/A-50
DOD B		220	97	1992,93	1095 (FOR 10 YR)	6000 LBS NTO/MMH
DOD C		400	65	1994	730 (FOR 8 YEARS)	7000 LBS NTO/MMH
SS CO-ORBIT PLATFORM	•	270	28.5	1997	365 (FOR 15 YR)	3300 LBS MMH/NTO
						330 LBS N2H4
EXPERIMENTAL GEO PLATFORM		19310	0	1999	730 (FOR 4 YR)	2200 LBS MONO/BI
SPACE ENERGY EXPERIMENT	•	270	28.5	1995	90	662 LBS MONO/BI
PLATFORM SYSTEMS TECHNOLOGY	•	270	28.5	1996,98	(1996,98)	2200 LBS MONO/BI
SOLAR TERRESTRIAL OBSERVATORY					30	AR,GN2
SOLAR TERRESTRIAL POLAR PLATFORM				1994	730	AR,XE,GN2

Figure 2.1-1 - Potential Satellite Propellant Resupply at Space Station



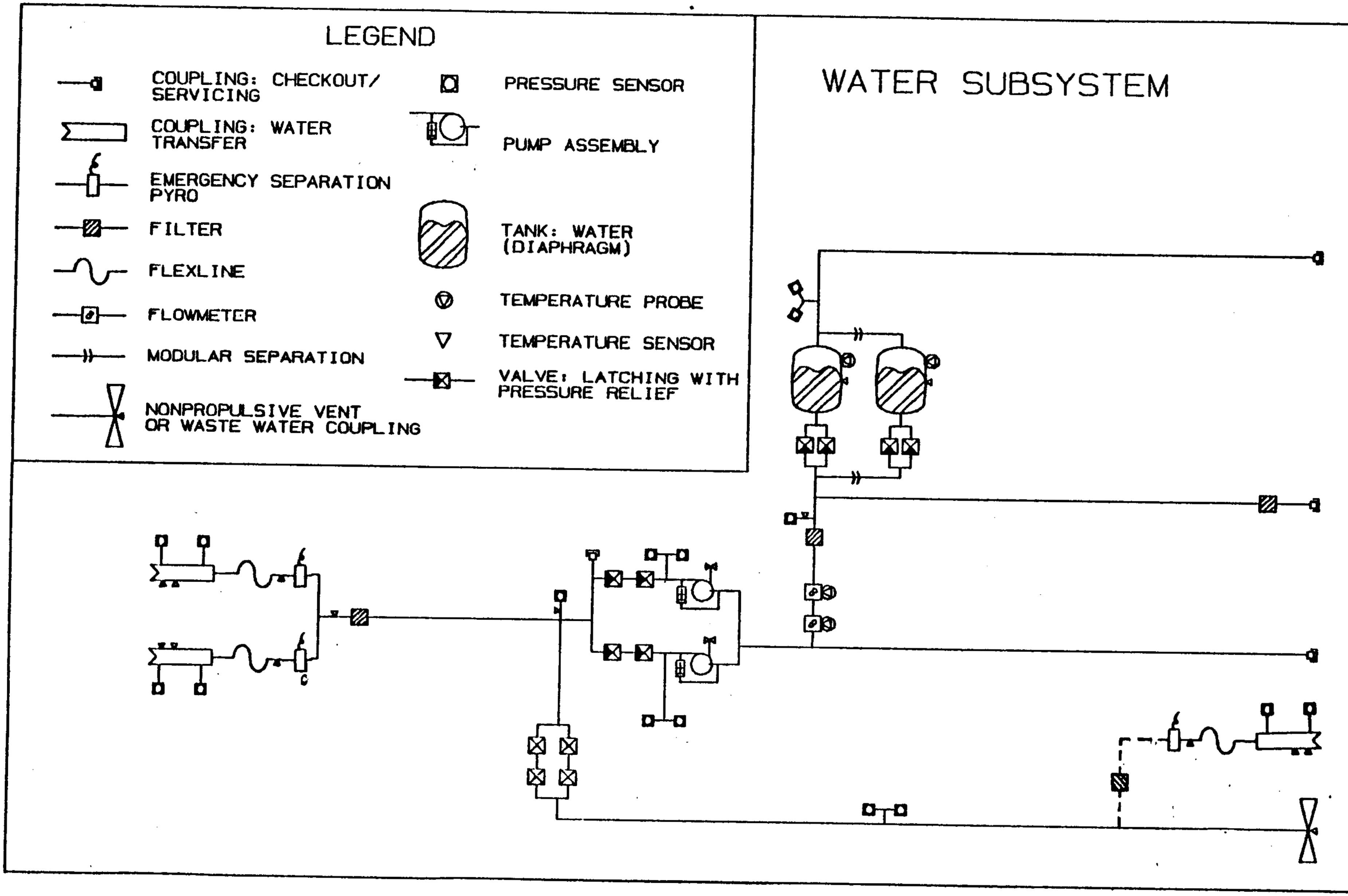


Figure 2.2-1 - Water Module Schematic

The required resupply quantity of water per year is determined by the difference between equivalent mass of water available and equivalent mass of water required. Available sources include waste gases and fluids, and recycled water contributions (about 8,025 lb). Required water sources include propulsion and lab requirements (about 14,065 lbs). Total resupply requirement is determined to be about 6,040 lbs per year.

2.3 Automated versus Crew EVA Functions

An early study indicated that successful future on-orbit resupply concepts preclude the use of EVA for any non-contingency interface function. The baseline OSCRS design lends itself to adaptation for remote resupply operations with minimum impact to planned structure. A modified standard end effector (SEE) will be used for berthing and partial rigidization. The SEE will be located in the center bay of the OSCRS surrounded by concentric fluid and electrical connectors on one modularized umbilical plate to facilitate remote connections. A single or small group of standard umbilical plates should be provided to potential OSCRS users to minimize impacts when switching resupply missions.

2.4 Optimization of Fluid Storage Capability

For consumables resupply using the OSCRS, the capability of a GRO sized tank for different liquid consumables are 1240 lbs of hydrazine, 1070 lbs of monomethyl hydrazine (MMH), 1780 lbs of nitrogen tetraoxide (NTO) (each filled to 93% usable capacity). Ullage bottles are used for the propellants to maintain a sufficient net positive suction head at the pump as a result of the larger delta pressures at the completion of resupply. For gas transfer, a set of six bottles with 12.5 inch outer diameter, 24 inch length were assumed, 20 lbs and 140 lbs of gaseous helium and nitrogen can be transferred per set, respectively.

Based on the annual projected consumable requirement (section 2.1): 3500 lbs of hydrazine, 3500 lbs of bipropellants, 6040 lbs of water, 120 lbs of gaseous helium and 840 lbs of gaseous nitrogen) the required liquid tank quantities are: 3 tanks of hydrazine, 4 tanks of bipropellants and 6 tanks of water. For gaseous consumables, 36 bottles are required for each of nitrogen and helium.

A tentative resupply schedule was evaluated assuming four resupply flights per year (Figure 2.4-1). A total of 17,280 lbs of consumables will be transferred by the OSCRS to the Space Station per year. Potential consumable quantity growth can be accommodated by the OSCRS tanker as indicated in Figure 2.4-2, where 6900 lbs of water and 420 lbs of gaseous nitrogen are carried in one flight. **Assumed at least two resupply tankers will be available and that each tanker will carry specific fluid.**

Figure 2.4-1 - Projected Consumable Resupply (Per Flight)

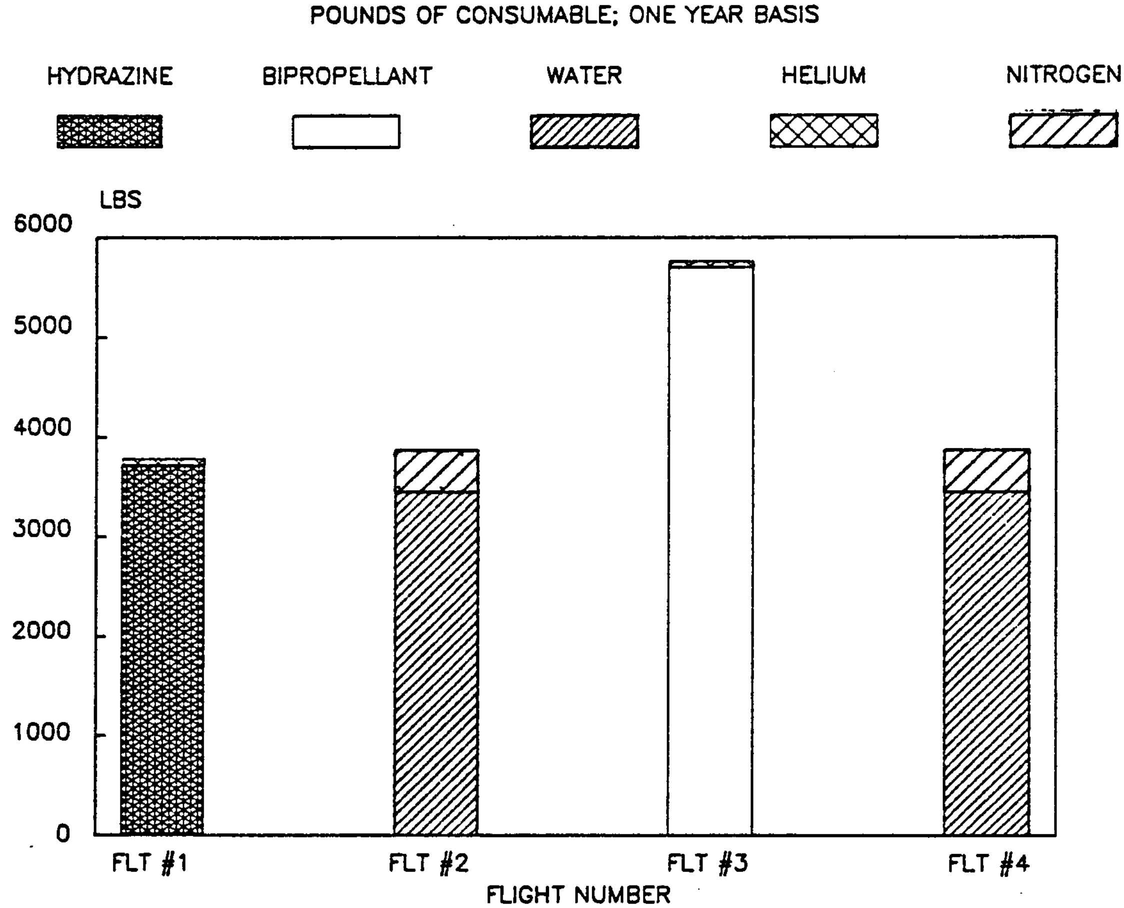
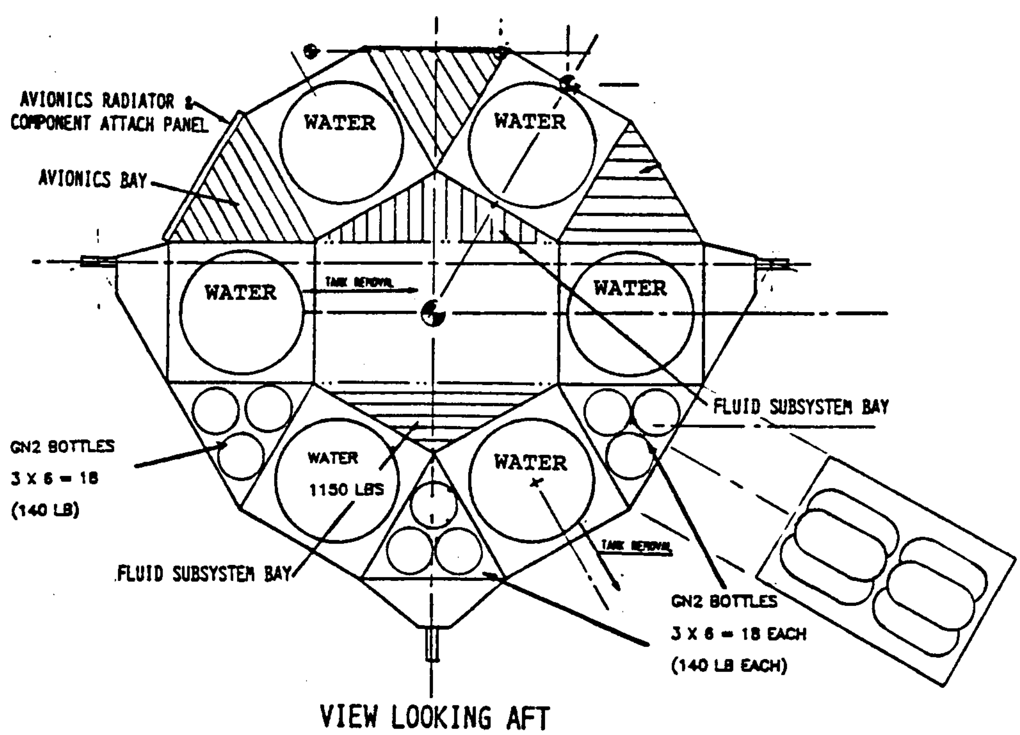


Figure 2.4-2 - Water Tank Arrangement in OSCRS



2.5 Offloading Tanker to Tanker

An application of the OSCRS tanker is to remain on-orbit at the Space Station to resupply the Station and other spacecraft. Potential weight and cost savings can be realized if the on-orbit tanker would offload residual propellant and pressurant to the replacement tanker. The study indicates that offloading pressurant gas could not be justified because the small weight of the pressurant to be offloaded would not offset the required additional weight to perform the fluid transfer. However offloading monopropellants and bipropellants can be justified by both weight and cost considerations. For monopropellants, the transferring of at least 350 lbs on the first flight will pay for the added components and launch cost of the new components. Subsequent flights would require the transfer of at least 140 lbs of the monopropellant. For bipropellants, the transferring of at least 445 lbs on the first flight and 155 lbs for each subsequent flight will justify the offloading of residual bipropellants from one OSCRS tanker to the replacement OSCRS tanker. The propellant offloading interface concept is shown in Figure 2.5-1 for the OSCRS tanker.

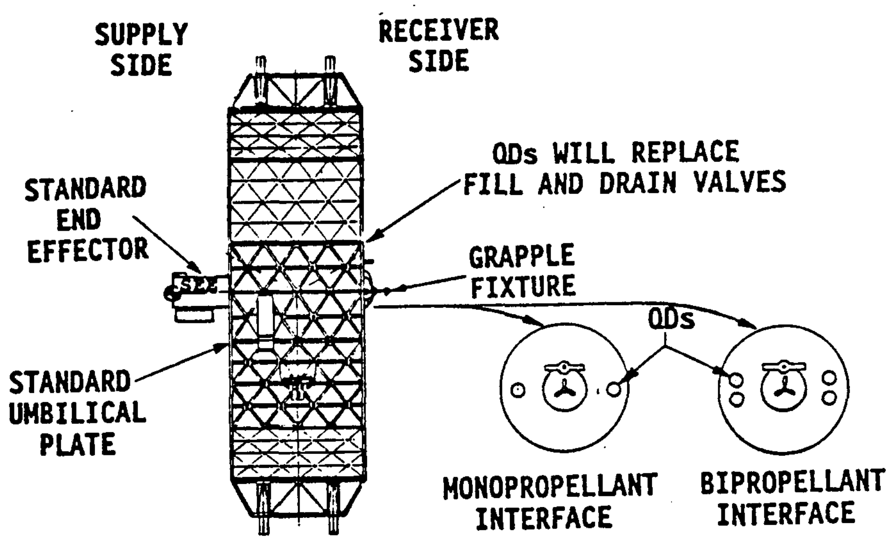


Figure 2.5-1 - Propellant Residual Offloading Interface Concept

2.6 Thermal Analysis

An analysis was completed for an OSCRS docked at space station to determine thermal and electrical power parameters. The earth-oriented station attitude results in selection of a continuous edge-to-earth OSCRS orientation (Figure 2.6-1). A 52-degree β angle, the station maximum, is used to try to generate a moderately cold condition. Shadowing by station elements was not considered.

Heater energy requirements are less during the first (fully loaded) half of each mission than during the second (depleted) half. This is partly due to the slight hot bias of the propellant tanks, which delays onset of the first heater cycle. The thermal control system is also more efficient when the tanks are fully loaded because the cooldown portion of the heater cycle is increased. The system may then reject heat at a slightly lower average temperature. The system warmup is rapid and the relatively higher average temperature, compared to that which occurs when the tanks are depleted, has little influence.

Maximum energy requirement for heaters plus avionics for the one week mission is 79 kwh when the avionics system operates at 380 watts continuously. For a continuous 140 watt avionics power dissipation, 44 kwh of total energy is required. Worst case heater energy for one week is 20.5 kwh, with 140 watts of avionics power dissipation.

A separate analysis of the avionics radiator showed that the addition of louvers does not result in temperature limit violations at 140 watts of avionics power dissipation. At 380 watts, the radiator can marginally tolerate direct solar heating. Combined earth and solar heating potentially could result in short term temperature limit violations. At higher avionics power levels, attitude restrictions are more severe. Because of the earth-fixed station attitude, preflight analysis and planning can avoid environmental heating extremes.

A representative case of OSCRS operation attached to the Orbital Maneuvering Vehicle (OMV) at space station and inclination has been analyzed. The flight profile is a seven-day mission at space station altitude to refuel a satellite and return. OMV capabilities, provided by TRW, for a worst-case orbit, are two 20-hour "active" attitudes separated by 90 hours dedicated to OMV battery charging, for a seven-day "subsatellite" mission. Thermal requirements of the OSCRS requires attitude 2 orientation during the specified mission. This was found to be an acceptable compromise between TRW-OMV and OSCRS requirements. The mission is illustrated in Figure 2.6-2.

The thermal model indicates that 19 Kwh are required for the nominal mission, reflecting a possible slight hot bias in the initial conditions. A total of 24 Kwh, twice the energy required for the second (depleted) half of the mission is a more conservative estimate. These results scale up to 21 Kwh to 27 Kwh for the total heater power requirement. Total electrical energy including avionics power is about 51 Kwh for the worst case.

Figure 2.6-1 - OSCRS Orientation Showing Fixed Attitude With Respect to Earth

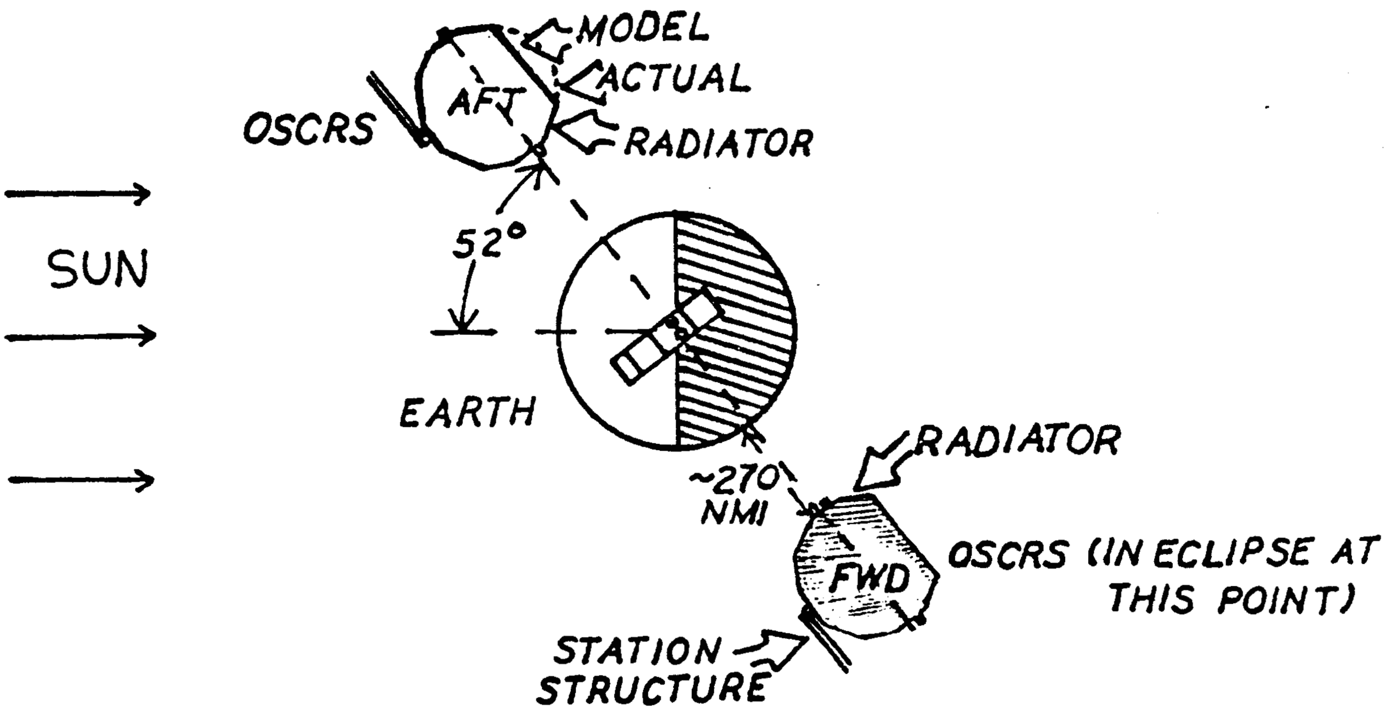
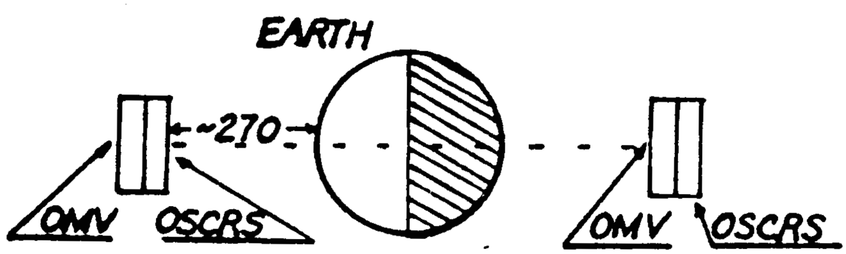
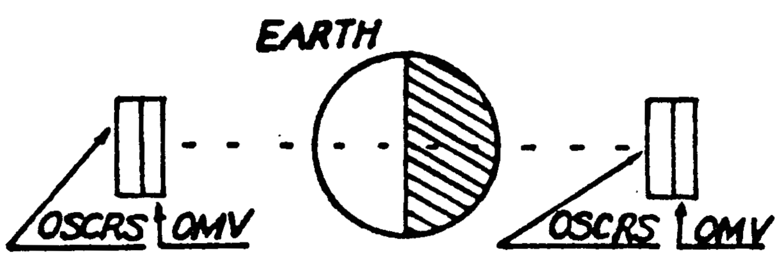


Figure 2.6-2 - OSCRS/OMV Mission Model



Attitude 1 Required by OMV



Attitude 2 Held Twice, 20 hr. Each, With 90 hr Gap in Between

The OSCRS thermal design must be modified for OMV long-term (7-day) operation to achieve power requirement limits (56.3 Kwh available from the Power Augmentation Kit plus 5 Kwh available from nominal OMV power). Modifications include increased solar absorptivity of OSCRS surfaces and radiator design changes. Weight penalty due to system modifications for this mode of operation is 16 lbs.

2.7 Operation of OSCRS at Space Station

Prior work on the OSCRS has been related to operation in the payload bay of the Space Shuttle. Operation at and attached to the Space Station necessitates a reevaluation of user requirements and the effects of those requirements on system operation, interfaces and software.

All of the previously defined requirements for refueling a receiving satellite must be met, not only using control from the aft flight deck (AFD) of the Orbiter but also from the control center of the Space Station. Interfaces between the OSCRS and the Station must replicate those in the Orbiter or the OSCRS adjusted to accommodate the Station interface.

Several interface approaches were considered. A simple replication of the AFD equipment in the Orbiter would require little redesign, however many wires carrying special functions for override and emergency operation are used. After consideration, it was concluded that a different approach to providing failure tolerance to the second failure and for emergency operation was appropriate. The control panel and computation system selected for the revised OSCRS which will provide the capability to operate from the payload bay of the Orbiter and the Space Station is a four string system. In the Orbiter, four control panel sections similar to the three AFD panel sections originally recommended for operation in the payload bay provide control and display of the commanded function or functions (Figure 2.7-1). Continued use of the Grid computer or similar graphic display on the AFD will provide flexible displays for observation of multiple system functions. Caution and warning will remain as a separate tie to the Orbiter caution and warning system.

At the Space Station, the same interfaces will provide control and display of commanded sequence(s). Both function control and display and flexible system detailed displays could be provided on a standard Space Station 'Multipurpose Application Console' (Figure 2.7-2).

The approach chosen will provide operation in the payload bay and from the Station while offering single failure tolerance for completion of operations and two failure tolerance for safety without the necessity for operator intervention. While the operator may intervene, the system is capable of isolating the first failure to a string by string comparison and powering down the failed string so that if a second

Figure 2.7-1 - 4 String P/L Bay OSCRS

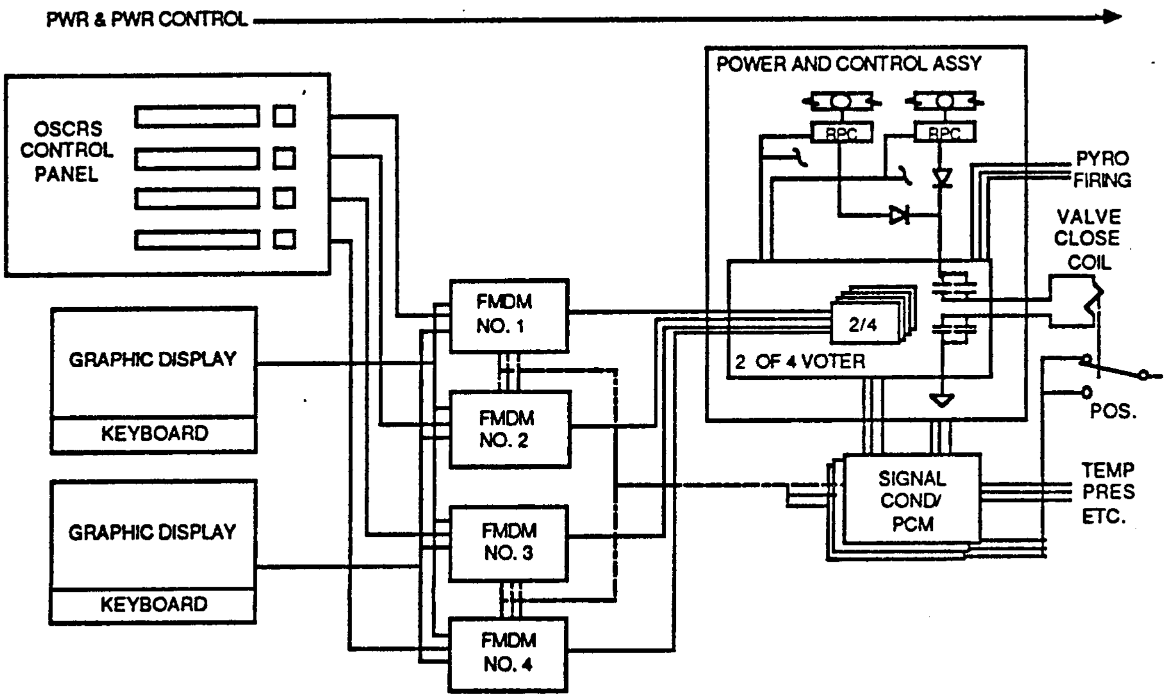
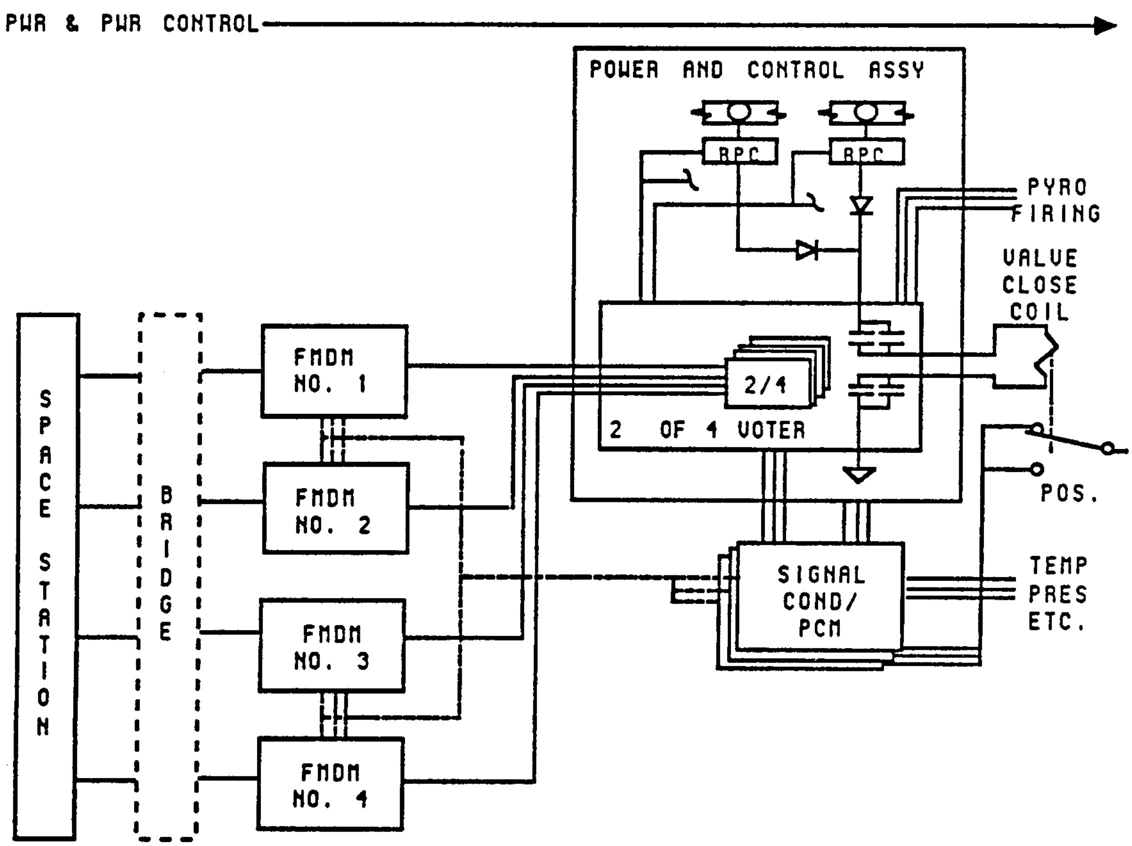


Figure 2.7-2 - 4 String Space Station OSCRS



failure should occur, safety will not be jeopardized. The operator will be notified of the second failure, at which time fuel transfer operations should be terminated and the system safed. While the system is still capable of fuel transfer operations, it is no longer failure tolerant and operations should be continued only for reasons strong enough to warrant the serious safety risks involved.

The probability of two simultaneous failures in the four FMDM's was considered to be very small.

2.8 Shuttle-to-Station Transfer of OSCRS Hardware

In transferring the OSCRS from the Orbiter payload bay to the Space Station for basing at the Station, it is desirable to maintain power to the heaters to prevent potential freezing of fluids. Depending on the fluids being carried, either freezing or thawing may cause damage to the fluid components. The development of this potential problem depends on the timelines required to transfer and the orientation of OSCRS to the thermal environment. The timelines are subject to cable connector removal and replacement times. Under reasonable assumptions the RMS transfer of OSCRS to the Station will take place without heating and instrumentation being required; but any problem in maintaining timelines indicates the necessity for both heater power and instrumentation to maintain and verify safe conditions during the transfer.

The OSCRS is basically in a quiescent state during the move so failure tolerance may not be deemed necessary; but if power supply circuit failure is to be at least one fault tolerant, two sources of power must be provided (present Space Station capabilities using the RMS is not fault tolerant). A second umbilical can be added to the Standard End Effector (SEE) or a battery subsystem added to OSCRS to support the transfer. The battery subsystem would be modular and installed as required for the transfer operation.

In a typical sequence, the SEE umbilical(s) would be connected to the OSCRS; then the Orbiter connectors would be removed; physical transfer would take place; the Space Station connectors would be connected and the SEE would be removed. Once the SEE is connected, the Station would be receiving data for monitoring the OSCRS as well as furnishing power to the heaters. Before the SEE umbilical is removed, the permanent Space Station connections would be in place. This suggested sequence would allow one failure to occur, while maintaining power and data flow to and from the OSCRS.

2.9 Central versus Multi Location Refueling Options at Space Station

The transferral of OSCRS from one location to a second location at the Space Station will create a similar situation to the Shuttle-to-Station transfer discussed in section 2.8. It was recommended that OSCRS is to remain "hooked-up" at all times; that one set of umbilicals be connected before the second set of umbilicals are removed. This approach will allow one failure to occur, while maintaining power and data flow to and from the OSCRS.

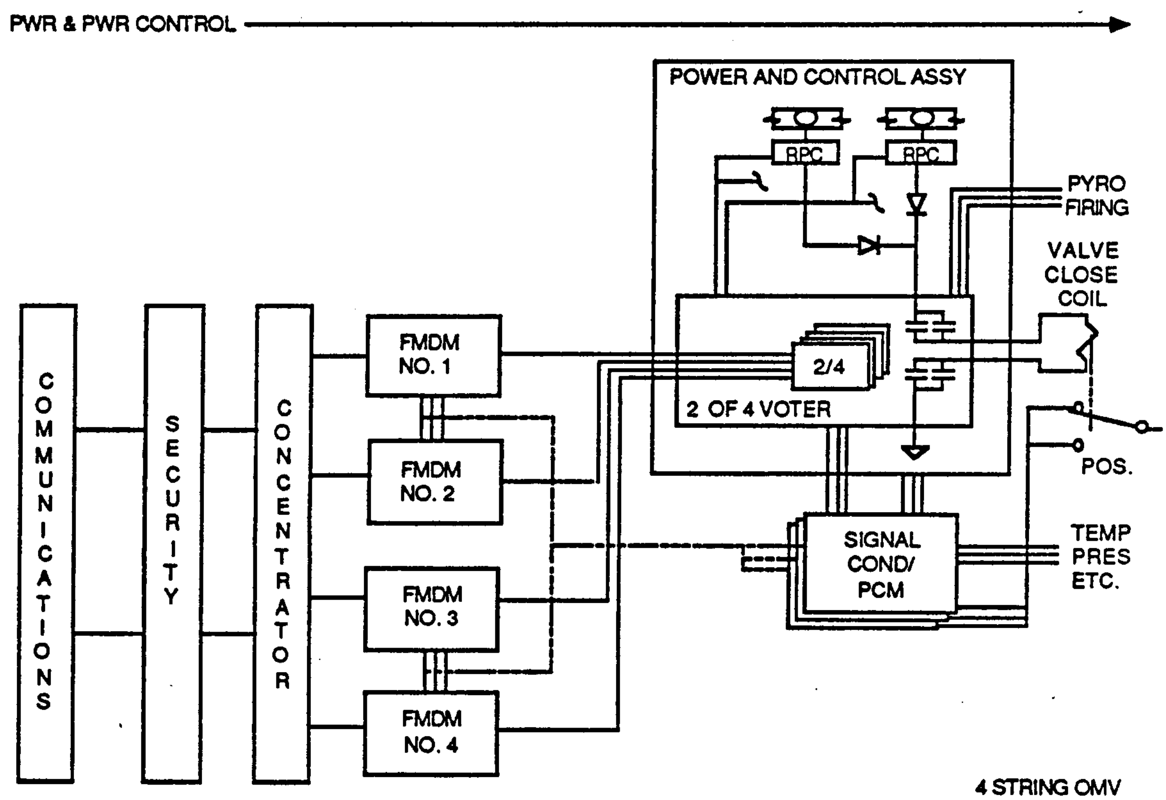
2.10 Operation of OSCRS Attached to OMV

The OSCRS must be able to transfer fuel while attached to the OMV as an OMV payload. This capability must include operation at or near the Space Station, near the Orbiter or at a remote location. In addition, the OSCRS/OMV must operate in a safe quiescent mode (no fuel transfer) while on an ELV during prelaunch, launch and on-orbit. It must also operate safely during transport in the Orbiter payload bay and while stored at the Space Station on or off the OMV.

In the present OSCRS design for Operation in the Space Shuttle, numerous functions are directly wired from the aft flight deck (AFD) to the OSCRS avionics. Included among these are the bank select power switching for fault isolation/fault effect avoidance, the safing circuits to override all other control in the event of massive failures, the control and sequence number feedback from the 'control panel', pyrotechnic actuation switches for emergency separation and power control for the FMDMs and heaters. To maintain the same type of backup while operating on the orbiting maneuvering vehicle (OMV), the separate circuits used in the orbiter would have to be 'recreated' in some way to be independent of the normal three redundant strings.

In order to operate OSCRS in any of the three environments (Orbiter payload bay, Space Station, or on OMV) it seemed prudent to attempt to arrive at a suitable common or similar interface between OSCRS and each of the three basing points. Several approaches were examined in a trade study performed to compare advantages and disadvantages of the various methods. A four string avionics system was selected because it provides the required safety and control survivability and is amenable to operation not only on the OMV but also in the Orbiter payload bay and at the Space Station. Four strings using standard protocols can be adapted to 'neck up' or 'neck down' at communications interfaces and thereby use available OMV channels as long as the requisite safety and survivability are assured. Figure 2.10-1 presents the four string OMV OSCRS concept.

Figure 2.10-1 - 4 String OMV OSCRS



2.11 OSCRS Launch Via ELV

Launching an OSCRS into LEO in support of supply/resupply missions will become a reality in the near future. STS orbiter cargo manifesting is and will continue to be at a premium. Relief must be forthcoming relative to OSCRS since frequent OSCRS launches will be a necessity. Launching an OSCRS into a LEO parking orbit on an ELV, rather than in the STS orbiter bay, was examined as a feasible concept.

The two primary parameters involved in determining the ELV launch requirements are payload weight and envelope. The selected launch vehicle is in production (i.e., Titan IV) and does possess an abundant payload weight capacity to launch an OSCRS/OMV into LEO. The Titan IV is capable of launching the OMV and OSCRS simultaneously. The combined weight of the OSCRS and OMV is 24,600 lbs and is less than the Titan IV payload launch capability of 64,600 lbs. The 200 inch diameter shroud is adequate for the OSCRS with minor modifications (local shroud pentrations). The OSCRS/OMV structural support system can be provided by two options using the Titan IV vehicle. The existing orbiter trunnion fittings on both OSCRS/OMV can be utilized by connecting each trunnion to the Titan payload support structure. The payload support structure may be independent of the existing shroud (Figure 2.11-1) or may include the shroud skin up to the shroud separation plane as shown in Figure 2.11-2.

The OSCRS ELV launch scenarios should include solo, mated to the OMV, and mixed cargo manifesting launch capabilities. The payload-booster interfaces will utilize the STS orbiter sill and keel trunnions integral to both the OSCRS and OMV vehicle basic structure. A solo launched OSCRS will require attitude stabilization after ELV separation. Concepts to provide on-orbit control while awaiting a rendezvous with the OMV were examined, but no final selection was made. A detailed analysis by the guidance, control and navigation specialists would determine the optimum stabilization system. This was not performed since it was out of scope of this study. The alternative to the OSCRS solo launch is a launch combining the OSCRS and OMV. Launching the OSCRS attached to an OMV has the advantage of using the attitude control subsystem provided by the OMV.

The launching of an OSCRS tanker, solo or in conjunction with the OMV and additional payloads, via an ELV is a valid scenario. Additional preliminary design and analysis remains; however, no major technical nor operational concerns have been identified.

SEPARATE ASE-SHROUD CONCEPT

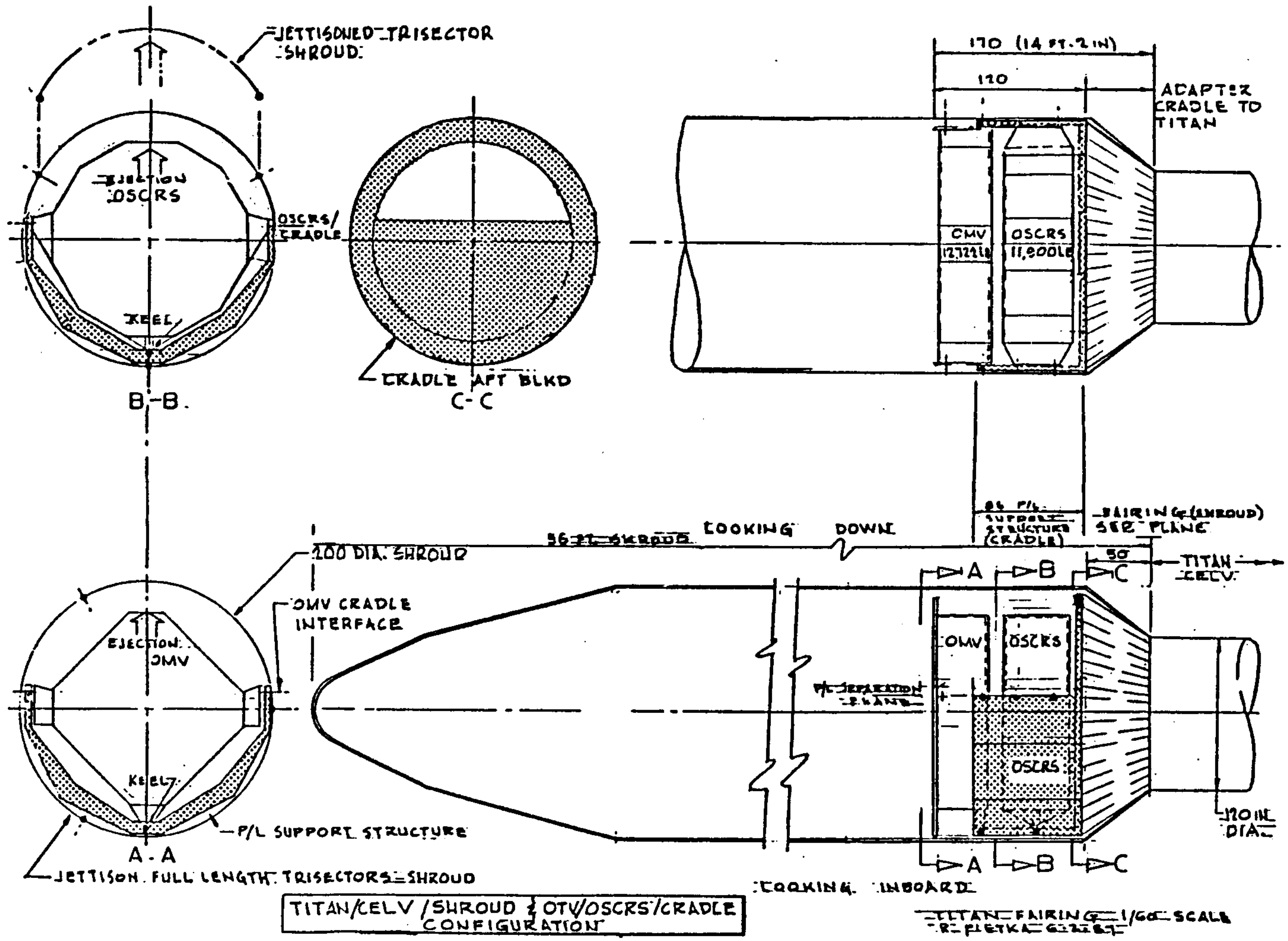


Figure 2.11-1 - ELV Launch Interface Configurations

INTERGRATED ASE-SHROUD CONCEPT

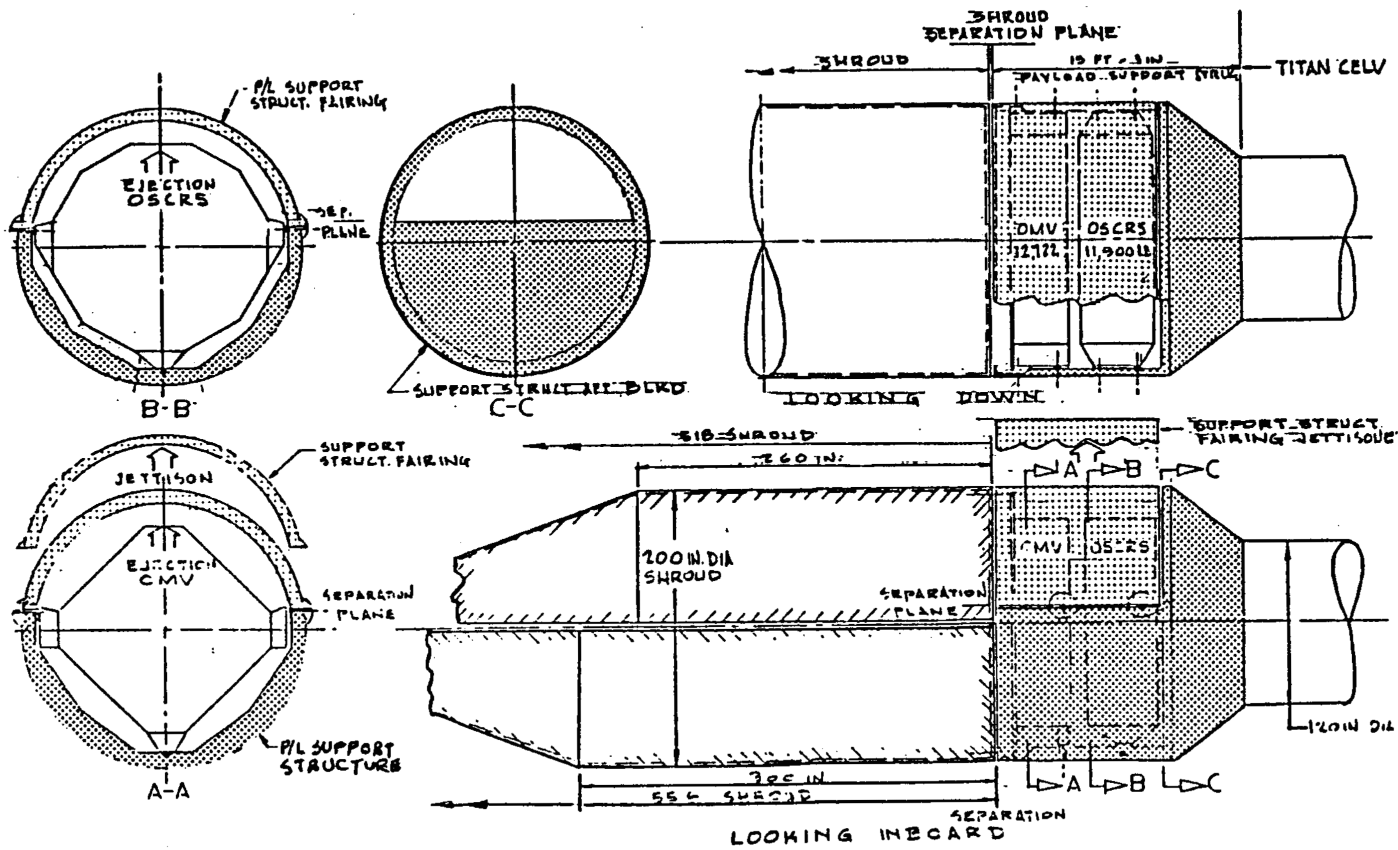


Figure 2.11-2 - ELV Launch Interface Configurations (Cont)

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3.0 Automatic Refueling Interface Design

A need exists for a NASA and industry standard umbilical interface in support of on-orbit consumables resupply missions. Considerable study, analysis, and layout effort was expended in determining the key requirements, solutions to operational problems and defining a standard on-orbit interface configuration. The result is a point design combining all the interface functions for spacecraft capture, rigidization, retrieval, umbilical engagement and contamination control in one unified, sequentially controlled operation.

An open truss structural configuration for the combined berthing/umbilical interface is shown in Figure 3.0-1 as installed in the current tanker concept. Calculated structural weight of the open truss support cylinder, support arms, shear panels, shear gussets, umbilical plate, attachment hardware, and a contingency factor is 170 lbs.

Several berthing/umbilical connector concepts were examined. The selected concept was developed by TRW-OMV/SPAR and has a linear travel actuator with a 26.6 inch axial travel capacity. This concept is favored since only the single TRW/SPAR linear actuator will be required to both capture/berth and engage the umbilical connector. However, a major drawback is to low axial load (pull-in) capability of 1200/1800 lb_f. This limitation requires an additional method of rigidization between SEE (standard end effector) and the FRGF (frangible grapple fixture).

The engagement forces result from coupling 8 fluid connectors, 4 gaseous connectors and 8 avionics connectors for a total engagement force of 1080 lbs. The number of connectors is a worst case scenario where the following types of connectors are required: two different propellants, ullage return for the two different propellants, two different gases and the required redundant connectors. The forces that occur during the transfer, 590 lbs at the fluid connectors, 585 lbs for the gaseous connectors, 800 lbs for the avionics connectors results in a total axial (separation) force of 1975 lbs. Applying a safety factor of 2 gives about a 4000 lb load requirement.

Figure 3.0-2 illustrates the additional rigidizing concept that is simple, lightweight and cost effective. Utilizing three pivoting arms located at 120° intervals around the outside circumference of the "SEE", it relies on redundant torsion springs acting about the arm pivot point to maintain the arm in the open position. This allows the FRGF to be drawn into the "rigidized" position by the snare carriage drive. As the translation drive retracts the "SEE", drawing the spacecraft to OSCRS, the pivot rigidizing arm is forced closed over the FRGF base plate creating a vise like grip or "super rigidization" of the "SEE"/FRGF interface.

The standard berthing/umbilical plate configuration is shown in Figure 3.0-3. The umbilical positions include 20 positions in two concentric rings surrounding the SEE assembly.

Figure 3.0-1 - Combined Berthing/Umbilical Structural Configuration

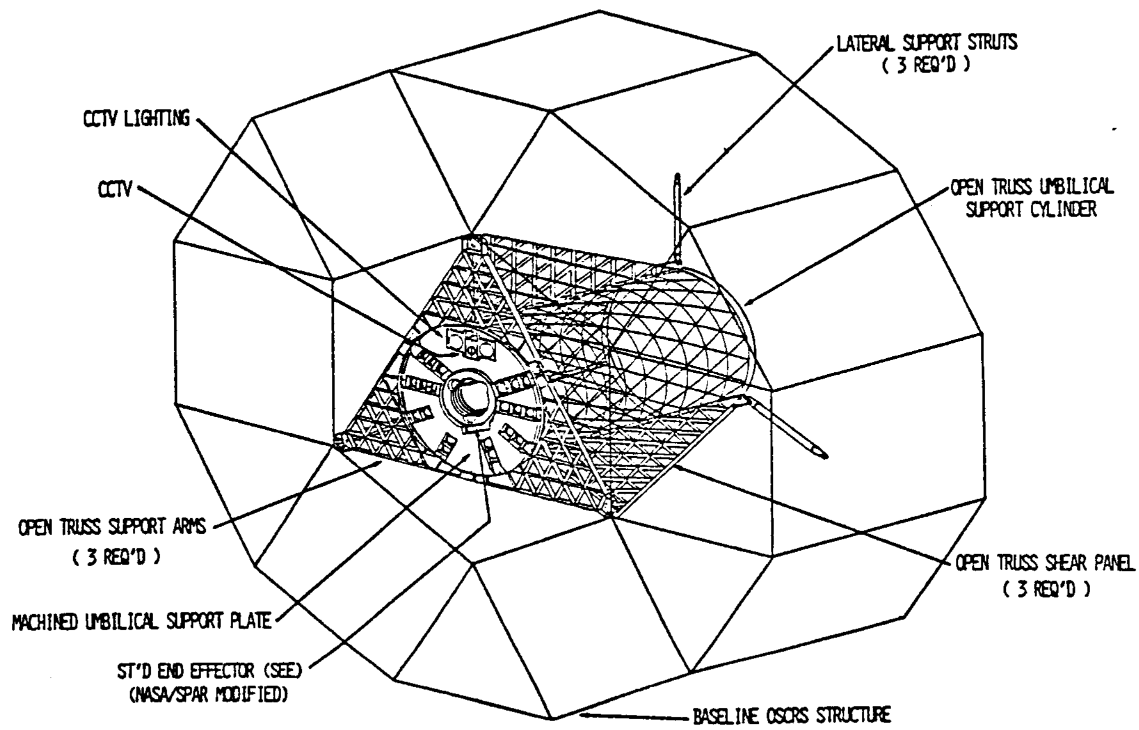
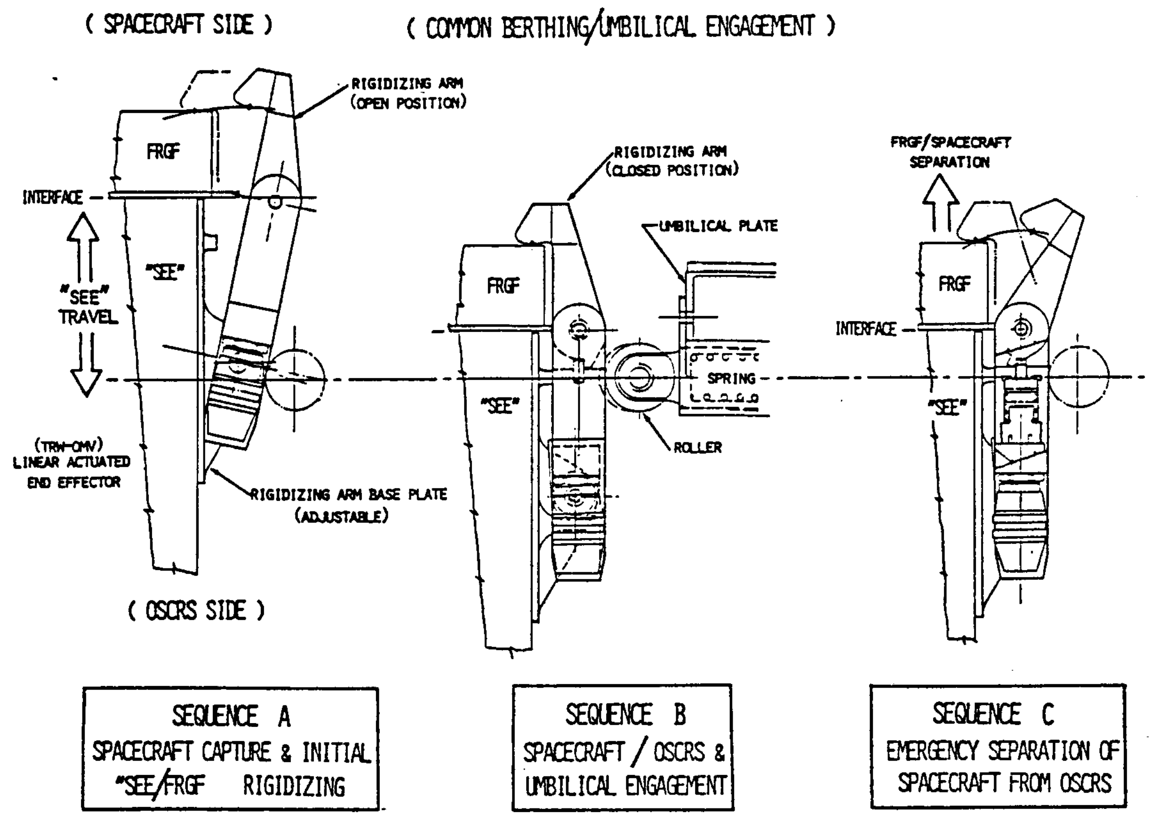


Figure 3.0-2 - Rigidizing Concept



To provide for contamination control and safety the umbilical requirements include: maximum separation between connnectors transferring different fluids, consistent with the limited space available on many spacecraft; staggered spacecraft connectors (for the different fluids) resulting in sequential connector engagement/disengagement; and contamination covers for all connectors.

The covers in the point design actuate by sliding on the interface side of the umbilical plate. The covers are semi-circular ("C"-shaped) in shape with oversized clearance holes appropriately spaced to permit passage of the spacecraft connector for engagement to the OSCRS connector mounted under the umbilical plate. The rotation of both covers is SEE-cam actuated. The cam controls the timing of the covers to the travel of the disconnects.

After spacecraft capture and initial "SEE" rigidizing, the spacecraft is pulled toward the OSCRS umbilical interface. At a predetermined distance from the interface, rigidization occurs as the three rigidizing links pivot over the spacecraft grapple fixture locking it to the "SEE". Figure 3.0-4 sequentially illustrates the engage/disengage action. The upper connectors, 1B and 7A are fixed/mounted to the spacecraft in a staggered relation (axially) to each other. The lower connectors, 1D and 7C are mounted, in-plane to the OSCRS umbilical plate. Connector 1D is fixed and 7C is allowed to traverse axially through internal compliance.

The MOOG connector (model 50E 565 RSO) was selected as the closest to design requirements for a fully automatic/remote operation type connector. Modifications to this disconnect were made after detailed discussions with MOOG to place all compliance requirements on the OSCRS side. Figure 3.0-5 illustrates the model 50E 565 RSO disconnect and the sequential ball rotation before, during and after engagement.

Featuring existing qualified hardware, where applicable, a point design emerged imposing very modest complexities over previous less automated concepts and greatly reduced the overall area required from past studies. Figure 3.0-6 presents three possible mission peculiar umbilical connector arrangements based on this study's concept.

4.0 Program Cost Estimate

Cost analyses were performed for two modules to be added to a monopropellant OSCRS tanker to perform a resupply mission in the mid 1990's. The first module costed is the water module designed to resupply water to the Space Station. The second module is an automatic/remote umbilical interface to allow fluid resupply without EVA activity. A cost analysis for a water tanker was determined with the inclusion of the two modules.

Figure 3.0-3 - NASA Standard Berthing Umbilical Plate Configuration

(ACCEPTS ALL FLUID AND GASEOUS TRANSFER CONNECTORS)

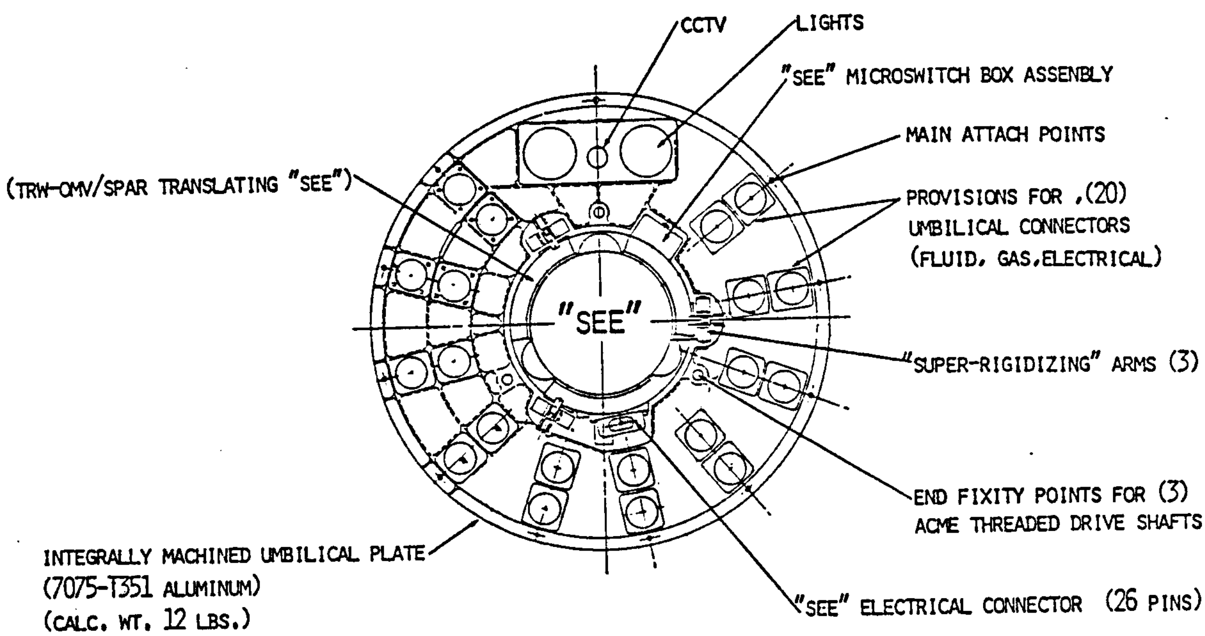


Figure 3.0-4 - Umbilical Engagement Sequence

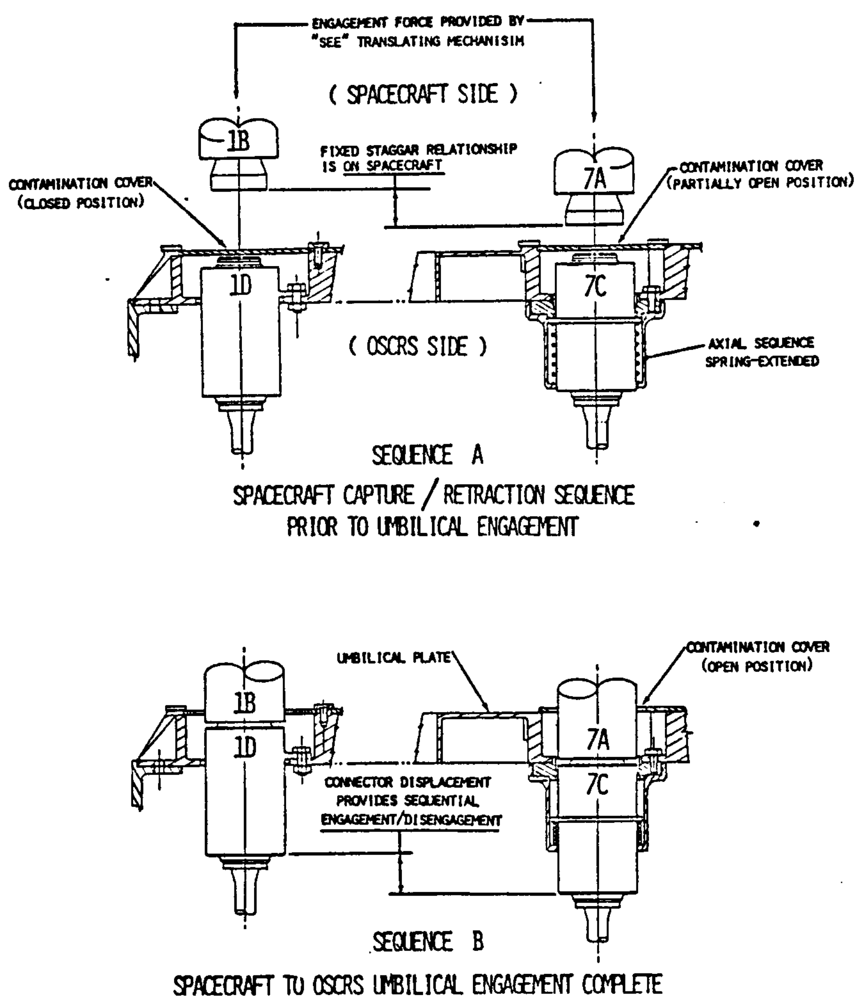


Figure 3.0-5 - Automated Connector Engage/Disengage Sequence

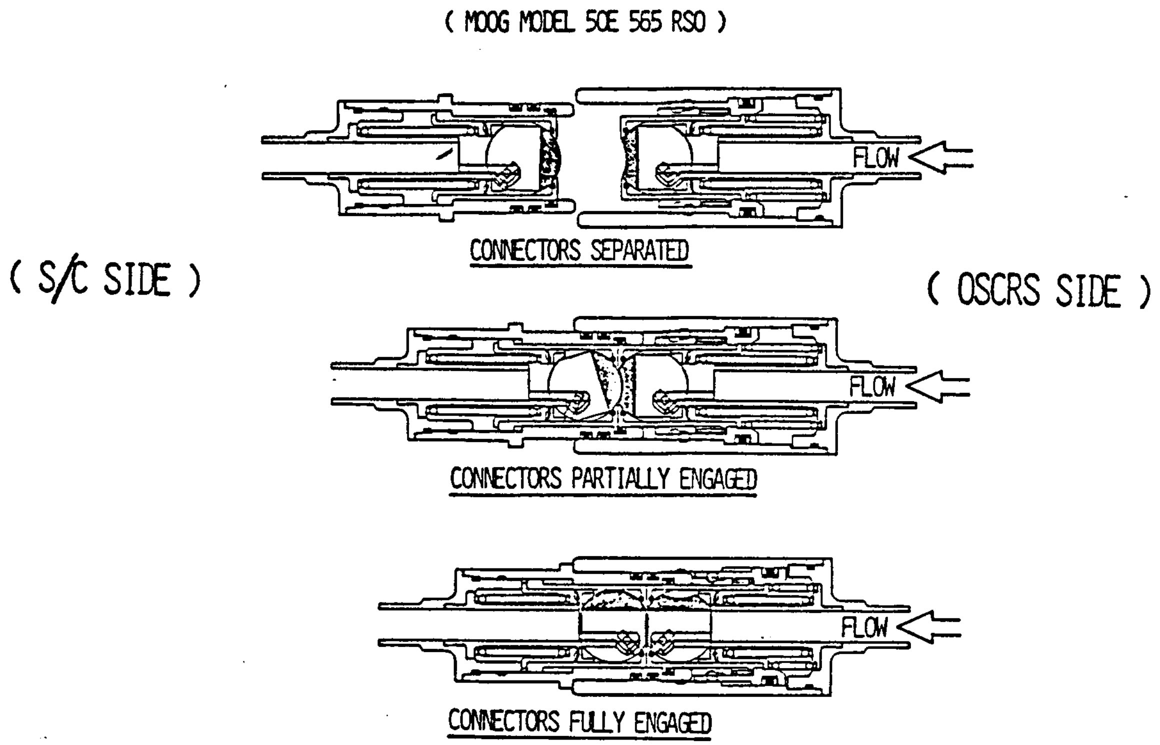
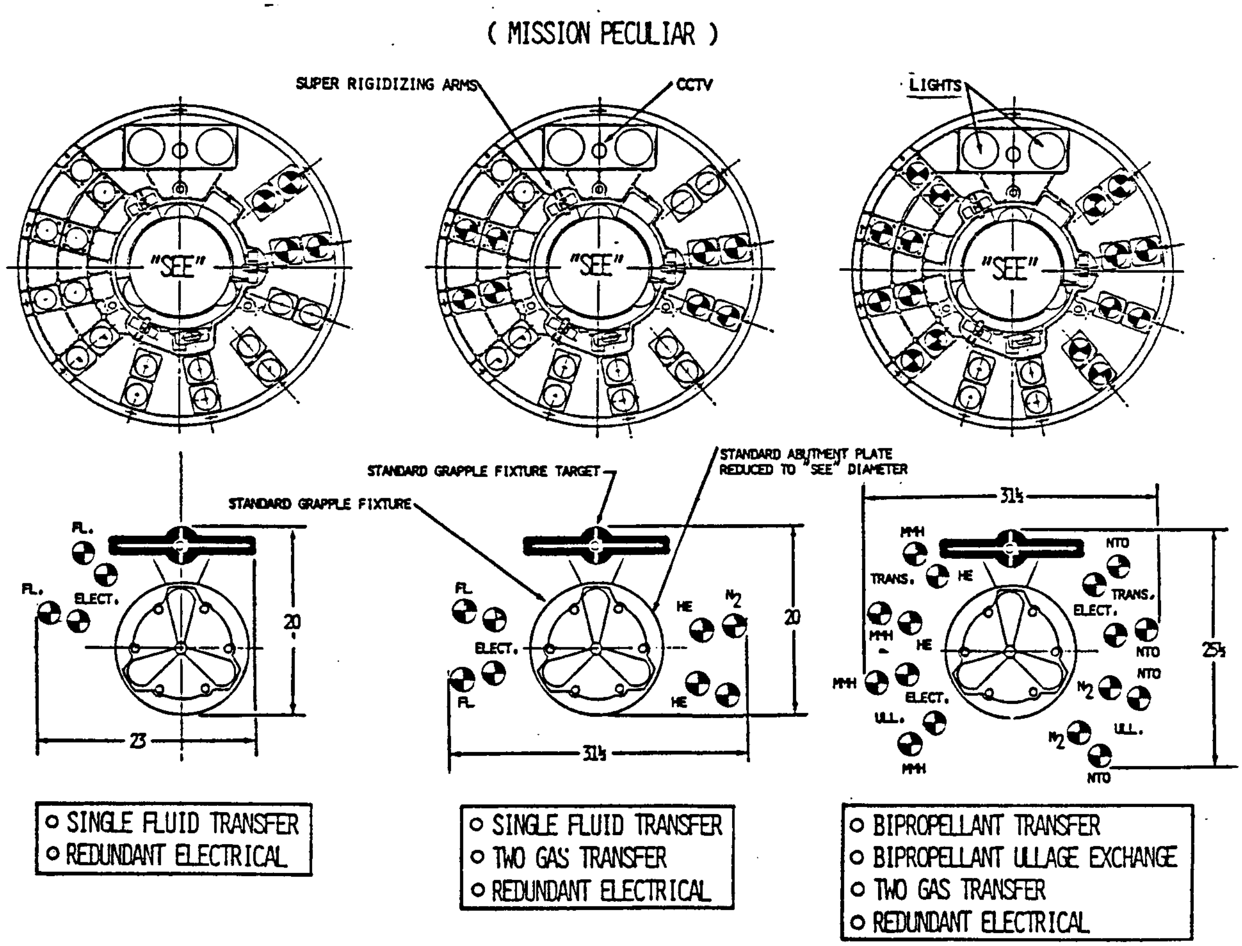


Figure 3.0-6 - Mission Peculiar OSCRS/Spacecraft Interfaces



4.1 Cost Optimization Efforts

The basic OSCRS tanker philosophy maximizes commonality between the baseline monopropellant tanker required to resupply GRO with 2500 lbm of N_2H_4 and future earth storable fluids resupply tankers which will be required to resupply over 7000 lbm of fluids. This commonality was optimized by use of the hybrid tanker concept which has a common structure for all earth storable propellant tankers, and modularizes all subsystems so that only mission essential elements need to be certified in the baseline tanker and flown in any future mission scenarios. **It was assumed that the costing for this tanker is based on the monopropellant tanker being built one year earlier.**

The modules are designed to replace existing modules (e.g. water subsystem for the hydrazine subsystem) or to be placed in predetermined structural regions. The structure is machined open grid aluminum alloy capable of holding six GRO size propellant tanks, and contains sufficient growth space for pressurant tanks as well as space for the control avionics to support these unspecified mission requirements. The tanker design, development, and fabrication will be of a configuration which includes 6 propellant tanks, space to add a pressurant resupply module, an ullage return module (as required), and the associated avionics and thermal control system.

4.2 Water Subsystem Estimated Cost

The cost to design, develop, and fabricate the water subsystem was estimated by engineering. The component costs are based on an analogy comparison to the baseline monopropellant OSCRS cost estimate, Rockwell costing for a water subsystem in the Space Station proposal, and selected subcontractor quotes.

Total nonrecurring costs are \$2.3 M, but a reduction in the nonrecurring costs will occur if Space Station and/or OMV component procurement precedes the OSCRS tanker component procurement. This would allow a nonrecurring cost reduction on several major components, i.e. the pump assembly, fluid coupling, water storage tanks, and the latch valves. Total recurring cost is \$6.0 M, with the major portion of the cost (\$5.0 M) for the six water tanks.

4.3 Automatic/Remote Umbilical Interface Estimated Cost

The total cost to design, develop, and fabricate the umbilical interface is \$9.0 M, with the nonrecurring and recurring costs estimated at \$4.2 M and \$4.8 M, respectively. The major recurring cost is the standard end effector at \$2.5 M.

4.4 Estimated Costs

The estimated cost for the water OSCRS through DDT&E and first unit production is $49.1 M. The total DDT&E cost with system engineering and program management is $22.1 million. Percentage breakdowns of DDT&E and production costs for the first deliverable system is graphically presented in Figures 4.4-1 and 4.4-2, respectively.

5.0 Executive Summary Conclusions and Recommendations

The study resulted in some significant conclusions and recommendations which should aid the NASA in directing the OSCRS program objective.

5.1 Significant Conclusions

- **Space Station resupply of hydrazine, MMH and NTO, and water should be provided by separate dedicated tankers. Station based tankers will resupply the Station, OMV and other spacecraft.**
- A hybrid-generic tanker can be economically developed to meet the resupply requirements for the next decade.
  - Structure is sized for over 7000 lbs of fluids
  - Modularization of subsystem elements limits scar weights to mission requirements only.
- Offloading monopropellants and bipropellants (old tanker to replacement tanker at the Station) can be justified by both weight and cost considerations.
- Launching an OSCRS into a LEO parking orbit via an expendable launch vehicle (ELV) was determined to be a feasible concept and this will help relieve premium STS orbiter cargo manifesting.
- An industry standard umbilical interface was designed to combine all the interface functions for spacecraft capture, rigidization, retrieval, umbilical engagement and contamination control in one unified, sequentially controlled operation.
- A four string avionics system will provide automatic faultdown through two failures in a consistent manner and its ability to 'neck-up/neck-down' at a data interface makes the system more adaptable to use with OMV and the Space Station.
- It will take approximately $49 M and a 41 month lead time to design, develop, qualify, produce and deliver a dedicated Space Station based tanker with a water subsystem and a remote/automatic interface.

Figure 4.4-1 - Breakdown of DDT&E Costs

TOTAL DDT&E COSTS = $ 22.1 M

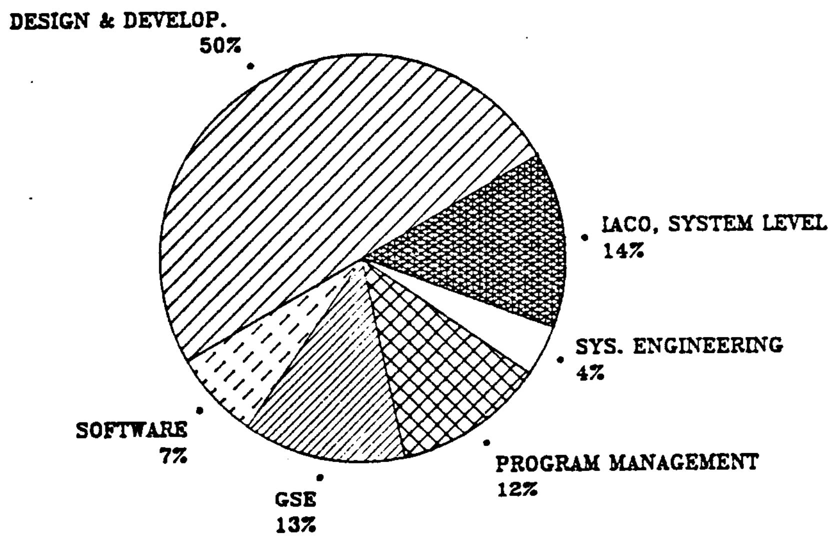
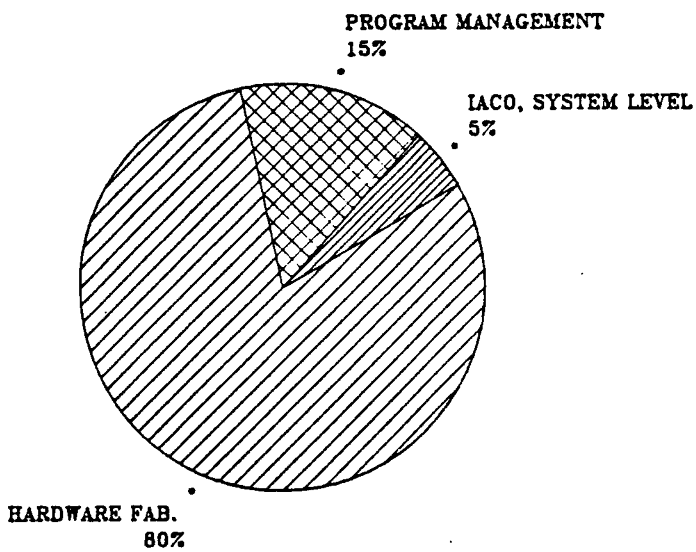


Figure 4.4-2 - Breakdown of Production Unit Costs

TOTAL PRODUCTION UNIT COSTS = $27.0 M



5.2 Recommendations

- To develop at least two generic tankers to be based at the Space Station.
- Develop a standardized remote/automated umbilical interface.
- Plan \$49 M and a 41 month leadtime to develop, qualify and deliver the second Space Station based tanker.
- Develop the capability to launch an OSCRS via ELV.

STS 86-0302-4

ORBITAL SPACECRAFT CONSUMABLES RESUPPLY SYSTEM (OSCRS)

FINAL REPORT
VOLUME IV
EXTENDED STUDY RESULTS

(DRD-10)

PART II
CONDENSED STUDY RESULTS

Prepared for
the
National Aeronautics and Space Administration
Lyndon B. Johnson Space Center

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CDRL DATA ITEM MA-1023T

SEPTEMBER 1987

Rockwell International
Space Transportation
Systems Division

CONDENSED STUDY RESULTS

6.0 Study Results Introduction

This report summarizes the study results of the Orbital Spacecraft Consumables Resupply System (OSCRS) study performed by Rockwell International for the National Aeronautics and Space Administration (NASA) at Johnson Space Center (JSC) under contract NAS9-17584. The study was performed in accordance with the study plan contained in modification 5C to the contract NAS9-17584. The study plan was set up to follow the Contract Change Authorization (CCA) SOW which expanded the basic contract SOW subtasks 1.1, 1.2, 1.3, 2.1.1, 2.1.2, 2.1.3, 2.2, 2.3 and 4.1 as shown in the schedule depicted in Figure 6.0-1. More detailed information on the study results can be found in DRD-6.

The objectives of this extended study consist of three major tasks. The first task is to establish the definition of Space Station and OMV user requirements and interfaces and to evaluate system requirements of a water tanker to be used at the Station. The second task is to conduct trade studies of system requirements, hardware/software and operations to evaluate the effect of automatic operation at the Station or remote from the Station in consonance with the OMV. From the results of the trade studies, establish the Station/OMV/OSCRS interfaces in an interface control document (ICD), establish a revised phase C/D program schedule and estimate phase C/D costs to incorporate unique Station/OMV features into the OSCRS. The last task is to evaluate automatic refueling concepts for use in the Orbiter, at the Space Station or remotely with the OMV and to evaluate the impact to OSCRS concept/design to use expendable launch vehicles (ELV's) to place the tanker into orbit.

6.1 Space Station Unique User Requirements

The resupply quantities and schedules for storable propellants for all OSCRS missions and potential space station resupply missions are listed in Table 6.1-2. The average yearly space station fluid resupply quantities are summarized in Table 6.1-1.

TABLE 6.1-1. SPACE STATION YEARLY AVERAGED RESUPPLY QUANTITIES

Fluid	Quantity (LBM)	Resupplied To
Hydrazine (N2H4)	3500	Spacecraft, OMV
Bipropellant	3500	Spacecraft
Water	6040	Space Station
GN2	840	OMV, Spacecraft
GHe	100	Spacecraft

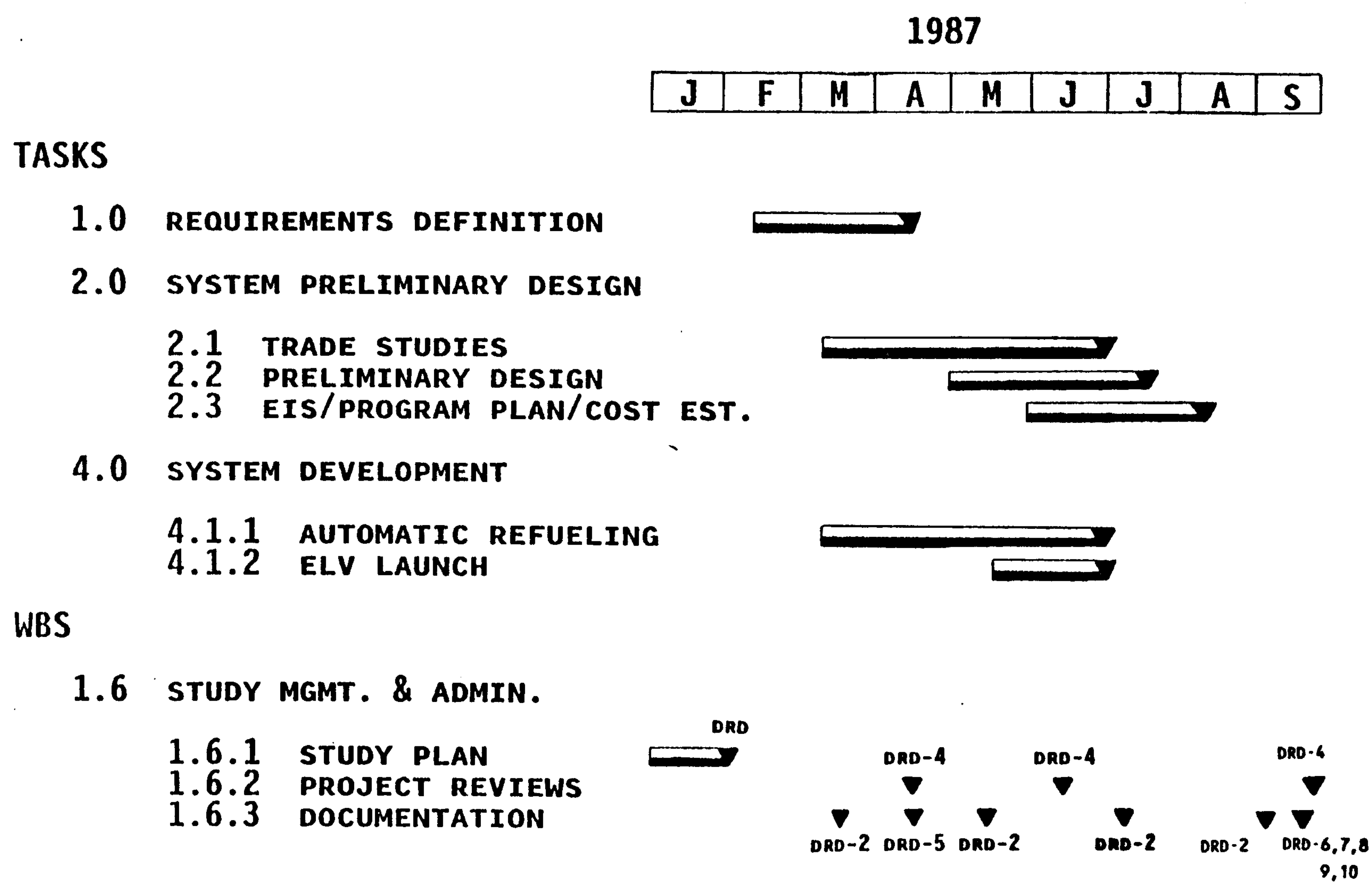


Figure 6.0-1 - OSCRS Study Extension Task Schedule

MISSION NAME	POTENTIAL SS RESUPPLY	ALT (NMI)	INC. (DEG)	LAUNCH DATE	RESUPPLY SCHEDULE (DAYS)	QUANTITY
GRAVITATIONAL WAVE DETECTOR	*	220	28.5	2001	180 (FOR 2 YR)	66 LBS N2H4
GRO	*	200	28.5	1991	730	2480 LBS N2H4
M-SAT-B		19310	0	1994	(1994)	1100 LBS N2H4
M-SAT-C		19310	0	2000	(2002)	2200 LBS N2H4
EXPLORER PLATFORM	*	220	28.5	1995	365 (FOR 9 YR)	551 LBS N2H4
TOPEX		720	64	1996	(1999 & 2002)	300 LBS N2H4
EARTH OBSERVATION SYSTEM		445	98.7	1996	365 (FOR 7 YR)	1800 LBS N2H4
EARTH RESOURCES SATELLITE		381	98.7	1995,98,01	(1997,00,03)	200 LBS N2H4
ASTRONOMICAL PLATFORM	*	220	28.5	1997	180 (FOR 7 YR)	397 LBS N2H4
SPOT		450	98.7	1993	1460	500 LBS N2H4
DOD D		450	98.7	1991,92,93,94	1825 (FOR 10 YR)	70 LBS N2H4
NAVY REMOTE SENSING SYSTEM		450	98.7	1990	1095 (FOR 12 YR)	70 LBS N2H4
EURECA	*	270	28.5	1994	730 (FOR 8 YR)	700 LBS N2H4
DOD A		125	96.5	1992 (2)	730 (FOR 10 YR)	7000 LBS NTO/A-50
DOD B		220	97	1992,93	1095 (FOR 10 YR)	6000 LBS NTO/MMH
DOD C		400	65	1994	730 (FOR 8 YEARS)	7000 LBS NTO/MMH
SS CO-ORBIT PLATFORM	*	270	28.5	1997	365 (FOR 15 YR)	3300 LBS MMH/NTO
						330 LBS N2H4
EXPERIMENTAL GEO PLATFORM		19310	0	1999	730 (FOR 4 YR)	2200 LBS MONO/BI
SPACE ENERGY EXPERIMENT	*	270	28.5	1995	90	662 LBS MONO/BI
PLATFORM SYSTEMS TECHNOLOGY	*	270	28.5	1996,98	(1996,98)	2200 LBS MONO/BI
SOLAR TERRESTRIAL OBSERVATORY					30	AR,GN2
SOLAR TERRESTRIAL POLAR PLATFORM				1994	730	AR,XE,GN2

Table 6.1-2 - Storable Propellant Resupply Requirements

6.2 Water Tanker Subsystem Requirements

In determining the impact to the OSCRS tanker to resupply water to the Space Station the first step is to determine the yearly water resupply requirements.

The required resupply quantity of water per year is determined by the difference between equivalent mass of water available and equivalent mass of water required. Available sources include waste gases and fluids and recycled water contributions (about 8,025 lb). Required water sources include propulsion and lab requirements (about 14,065 lbs). Total resupply requirement is determined to be about 6,040 lbs per year.

Figure 6.2-1 represents the water subsystem schematic. The type of pump that is recommended is the centrifugal pump. The maximum pressure required in the SS water tanks will be about 50 to 200 psia. The centrifugal pump is the most efficient, lightest, and lowest power user of the types of pumps available for the anticipated water transfer. Power requirements are further reduced by blowdown to pressure equalization (between Space Station and Tanker) before pump activation.

The water subsystem is a simplified version of the hydrazine subsystem. Water is a benign fluid with an existing transfer technology base. This will allow simplified resupply and disposal interfaces (a reduction from 3 poppets to 2 poppets, and an increase in allowed fluid volume release at coupling separation). Also fewer components are required in the water subsystem. Table 6.2-1 presents the expected fluid subsystem weight of 298 lbm. The simpler water subsystem is about 161 lb lighter than the hydrazine subsystem with 2 propellant tanks (454 lbm).

Table 6.2-1 - Fluid Subsystem Weight in pounds

Component	Number	Weight/unit	Total Weight
GRO tanks	2	99	198
valves	12	2	24
flow meters	2	7	14
pump assembly	2	10	20
quick disconnect	2	5	10
pyro device	2	4	8
flex line	2	1.5	3
line	20 Ft	.25/ft	5
filters	3	1	3
fill and drain	1	2.2	2.2
non-propulsive vent	1	5	5
test ports	3	0.2	0.6
Total			292.8 lb

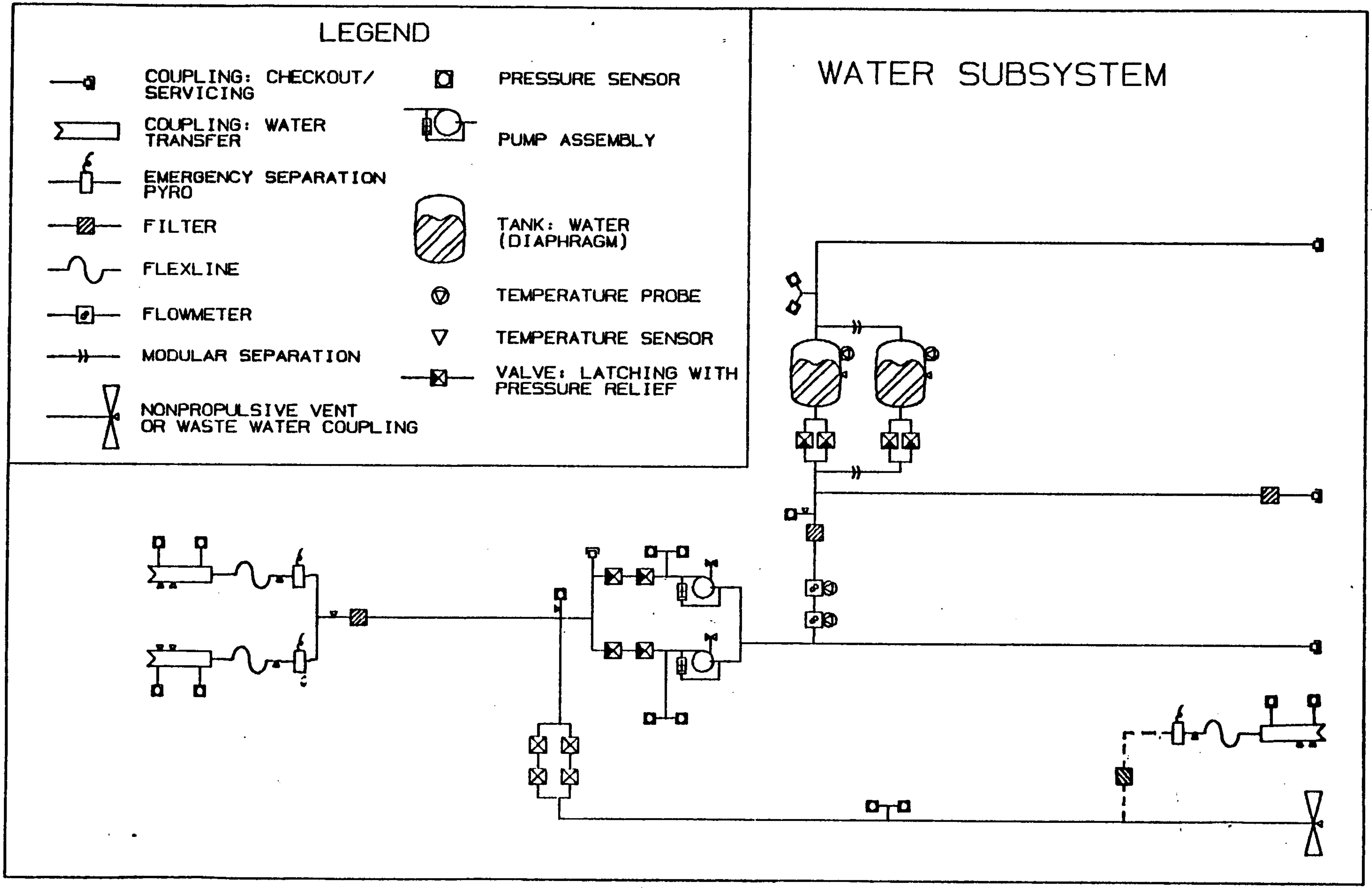


Figure 6.2-1 - Water Module Schematic

The primary structure used for the monopropellant and bipropellant subsystems will remain unchanged. The water system will be installed (and replace one propellant subsystem) with only minor changes expected in the secondary structure. Secondary structure changes will occur due to fewer fluid components (reduced subsystem weight) and differences at the fluid resupply and waste disposal interfaces.

As in the primary structure, the avionics subsystem will remain basically unchanged. The avionics subsystem is designed for complex propellant transfer and it can easily accommodate a simpler water transfer. Very minor modifications to the software are expected if any changes are required at all.

A reduction from 102 temperature sensors to 59 temperature sensors occurs when the water subsystem replaces the basic monopropellant subsystem. Reductions occur primarily in the fluid subsystem because of reductions in the number of fluid components and reduction of the number of sensors on each component. For example, a twenty sensor reduction occurs on the valves due to a change in fluid (hydrazine to water).

6.3 Automated Versus Crew EVA Functions

The future of successful space-based, (i.e., Space Station, OMV, etc.), consumables resupply rests with the timely development and standardization of automatic/remotely operated interfaces. Principally these interfaces include berthing (including interface rigidization, as required) and umbilical components (fluid and gas valves, and electrical connectors).

Berthing Interfaces

Operational scenarios must be constructed and evaluated in order to determine what, if any impact automatic/remote operations might have on the OSCRS baseline design. Figure 6.3-1 illustrates five such scenarios including the baseline OSCRS/orbiter resupply to the gamma ray observatory (GRO) vehicle (Figure 6.3-1(A)).

Figure 6.3-1(B) depicts OSCRS attached to both the CMV and a S/C. The initial interfaces are provided by modified standard end effectors (SEE-M). The modification adds a structure surrounding and attaching to the SEE. This permits the SEE to extend and retract along the SEE C axis. This SEE modification is presently integrated into the OMV baseline design.

Figures 6.3-1(c) and (e) are variations of the same initial interfaces described for Figure 6.3-1(B).

Figure 6.3-2 illustrates the modified SEE added to the OSCRS baseline structure with only minor structural modifications to allow the SEE to OSCRS attachment. If further rigidizing between S/C and OSCRS is required (if umbilical engagement/disengagement loads exceed the limits of the SEE) then MMS/FSS type latches may be added.

The specific operational load limits of the SEE are identified in Figure 6.3-3. The dimensional tolerance relationship between the SEE-M and a S/C after SEE-M rigidization is also shown. This dimensional relationship represents the possible C misalignment between fluid, gas and electrical disconnect halves. Lateral/radial compliance of the individual components must include this $\pm$ 0.1 inch dimension.

OMV Interfaces

Three major interfaces exist on the OMV for attachment to other S/C or Space Station:

1. Provisions to support cantilevered payloads weighing up to 10,000 lb-ft nominal/13,000 lb-ft maximum exist in the form of 8 attach point locations integral to OMV basic propulsion module (PM) structure.

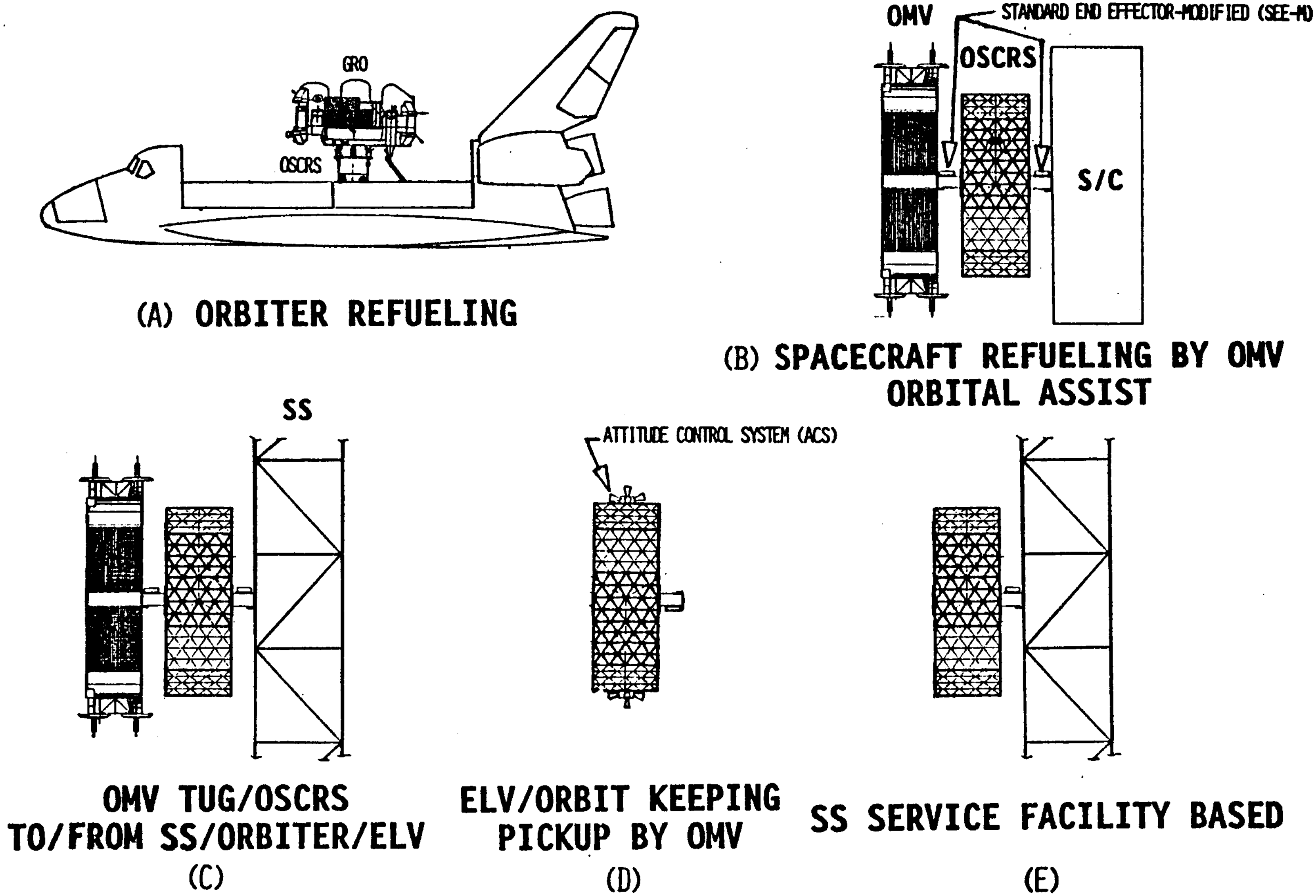


Figure 6.3-1 - OSCRS Operational Scenarios

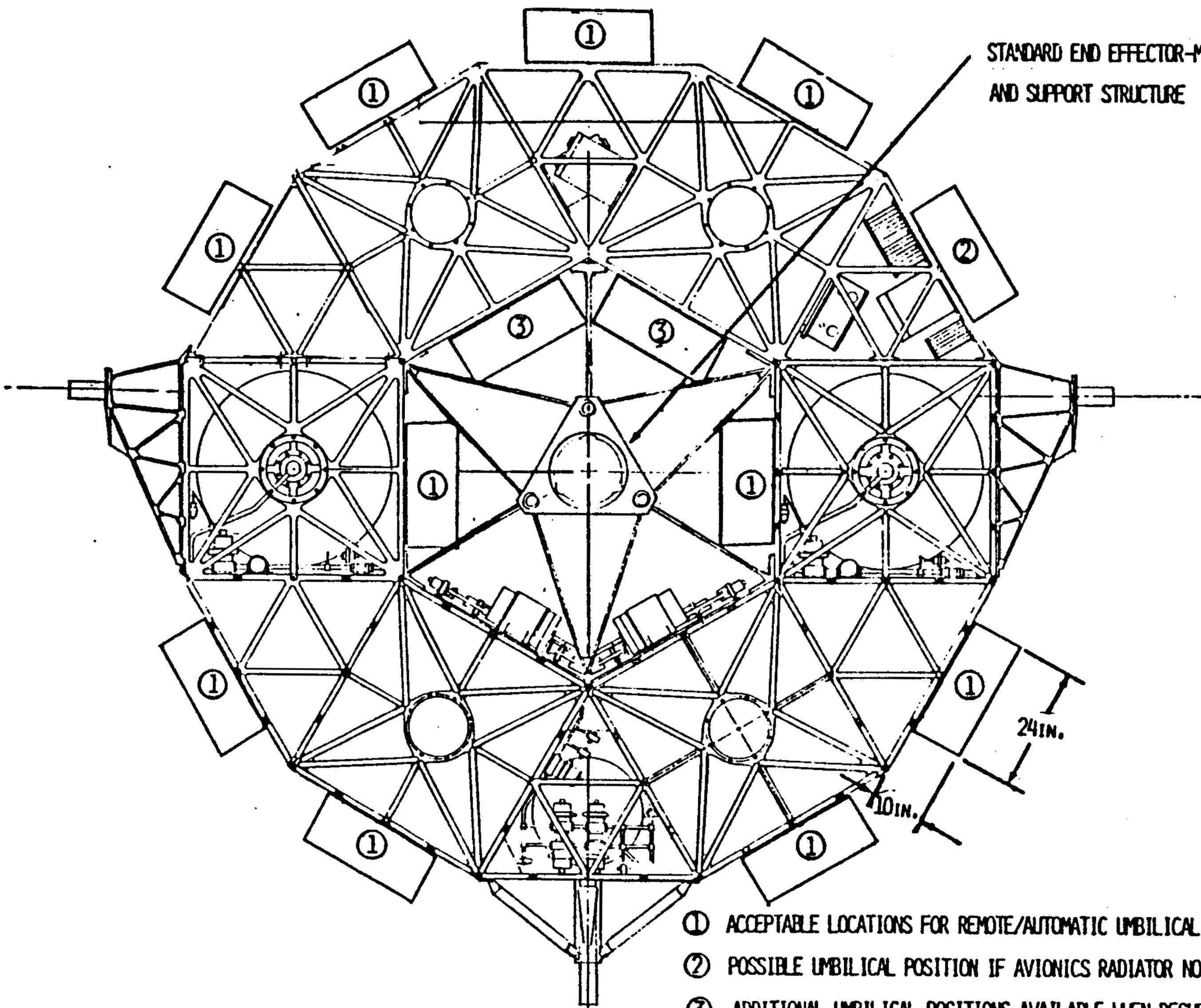
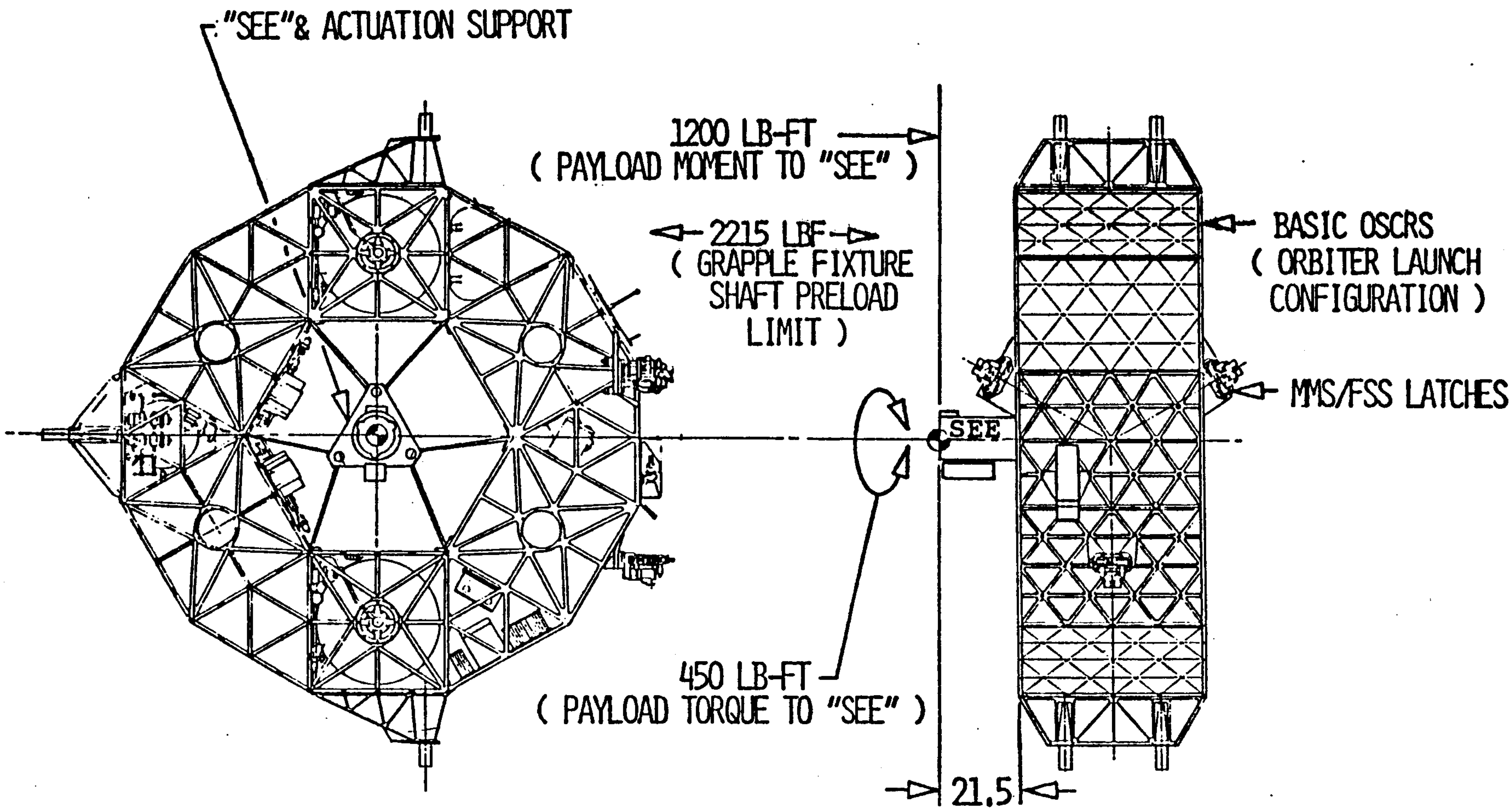


Figure 6.3-2 - OSCRS Baseline Provides Multiple Umbilical Locations



: AFTER SEE RIGIDIZING, RELATIONSHIP BETWEEN SEE AND PAYLOAD KNOWN WITHIN: ± 0.15
± 0.1 INCH

f_n (PAYLOAD/GRAPPLE COMBINATION): > 5 HZ

Figure 6.3-3 - Standard End Effector Docking Design Requirements

2. There are 4 interface attach points located on a 65.0 inch diameter.

3. A modified SEE-M is the third interface and has been described previously.

The baseline OSCRS design can adapt to the use of these three interfaces with minimal structural impact.

Umbilical Interfaces

The baseline OSCRS was designed to provide latitude in resupplying various liquids. The space station consumables requirements are presented in Table 6.3-1. Table 6.1-1 gives storable propellant resupply requirements for known S/C missions. With minor modifications, all of the media listed in these tables could be transferred utilizing a baseline OSCRS, although not on a single resupply mission. The modifications would be in seal materials and subsystem simplification for different liquids. A standardized remote/automatic umbilical system should be compatible with the transfer of the various fluids listed in Tables 6.3-1 and 6.1-1.

The following umbilical disconnect requirements were established:

1. Reliability/Complexity
2. Weight
3. Cost
4. Pressure Drop
5. Leakage (connected/disconnected)
6. Engagement/disengagement forces
7. Two-failure tolerant seals
8. Minimal leak paths
9. Ability for disengagement under pressure
10. Materials Compatibility

A survey of existing industry disconnects was conducted and evaluations made to determine if acceptable "shelf" hardware was available that met these requirements.

Based on the survey, the MOOG Model 50E 565 RSO, shown in Figure 6.3-4 was selected as possessing the characteristics best satisfying the operational requirements outlined.

The umbilical general arrangement shown in Figure 6.3-5, which consolidates a group of bipropellant, gaseous, and electrical disconnects, utilizes the MOOG disconnect as the "prime" component. The eight disconnects shown "group" on a 10 x 24 inch single umbilical plate. This particular arrangement does not provide for disconnect

Table 6.3-1 - Average Yearly Fluid Resupply Requirements at IOC Space Station (lbm)

FLUID	STATE	ECLSS	USL	JEM	PROPULSION & POWER	POTENTIAL OSCRS RESUPPLY
ARGON	GAS		167	609		776
PROPANE	GAS			13		-
CHLORINE	GAS			18		-
CARBON DIOXIDE	GAS		104	44		-
FREON	LIQ		3			-
HELIUM	GAS		9	18		27
HYDROGEN	GAS		3	1		-
KRYPTON	GAS			40		40
AMMONIA	GAS			4		-
NITROGEN	GAS	2028	808	54		2890
NITROGEN	LIQ		312			-
OXYGEN	GAS		122	30		-
SILICON HYDRIDE	GAS			13		-
WATER	LIQ		4541	335	2500 TO 16000	7376 TO 20876
XENON	GAS		44			44
CLEANING	LIQ		821			821

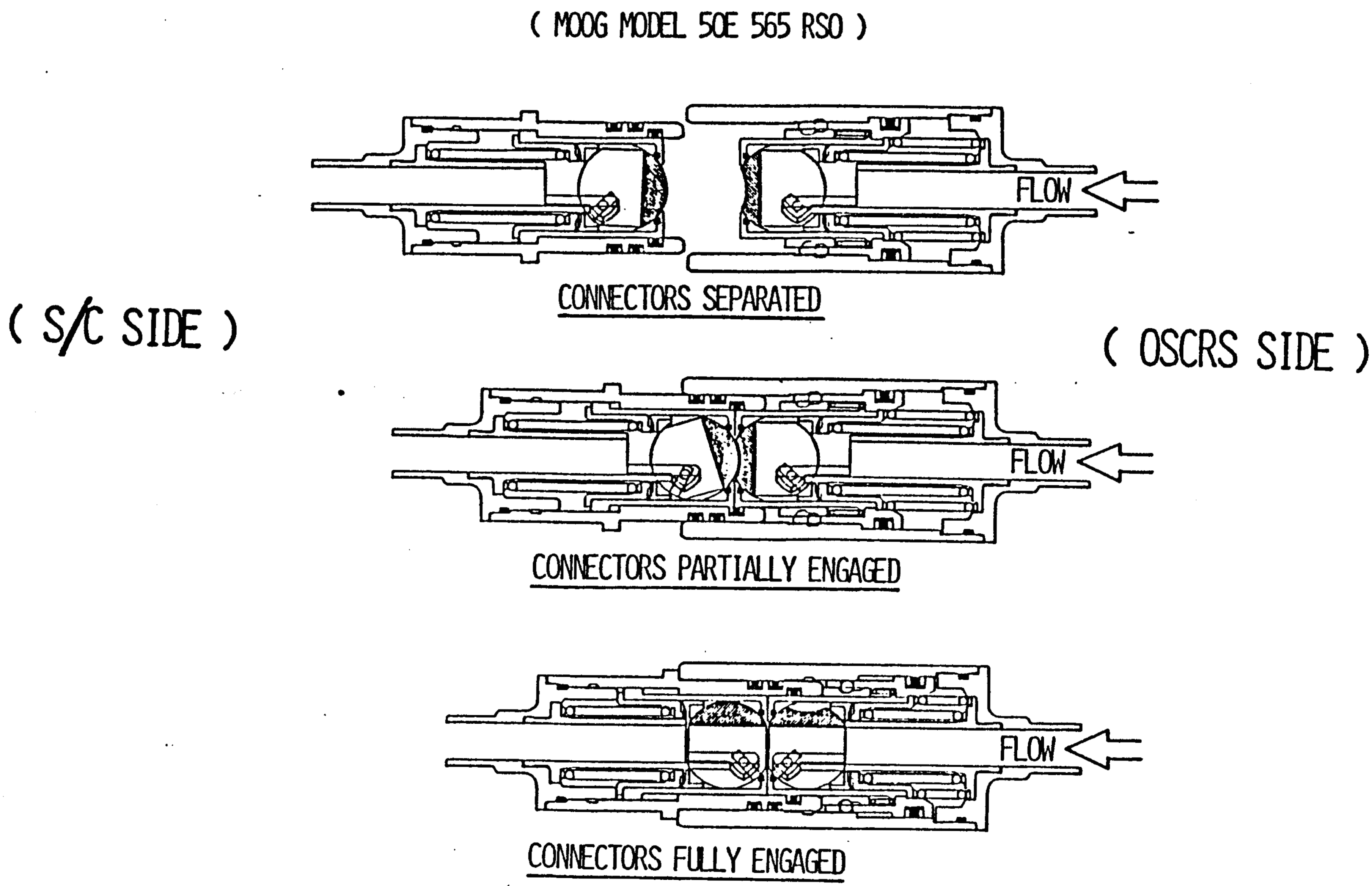


Figure 6.3-4 - Umbilical Engagement Sequence

redundancy. It does address safety issues regarding separation distances between hypergolic propellants. Figure 6.3-5 illustrates a minimum envelope to provide a "starting point" in developing a standardized umbilical interface design.

Space Station

Consumables transfer to Space Station involves the transfer of many fluid and gas types (Reference Table I). The external fluid/gas systems interfaces presently foreseen involve pressures from zero (venting) to 4,500 psi and line diameters ranging from 3/8 to 2.5 inches. The individual connector requirements approach 900 units in number. All units are manually operated (axial push-pull travel) and most are of the "quick-disconnect" type. All disconnects are self-sealing.

Serious consideration is being given to the use of robotics as the "manual" disconnect operator; however, the operating forces (requirements) will be limited to that available from individual EVA crewmembers.

Whether the present Space Station connector philosphy will dictate manual type connectors, therefore requiring "manual types" in addition to automatic connectors in remotely operated "gang" umbilical assemblies, remains an open issue.

Thermal Requirement

Various consumables require active thermally controlled environments; i.e., N_2H_4, N_2O_4,/MMH, and H_2O. Umbilical disconnects transferring these fluids must provide adequate thermal conditioning including across the interfaces.

It seems unreasonable and operationally defeating to isolate Space Station resupply interfaces, related to OSCRS, from other S/C remote/automatic interface requirements. It is recommended that all transfer interfaces involving OSCRS be designed for remote-automatic operations.

The baseline OSCRS appears to be very accommodating to remote/automatic umbilical and berthing requirements as defined without significant impact to basic structure or subsystem layouts.

It is recommended that, in addition to the requirements originally designed to, i.e., TRW/GRO, all future aspects of resupply with an OSCRS vehicle utilize a remote/automatic operation to the maximum extent possible. Only unplanned EVA (contingencies) and planned observations (surveillance) should be included in timeline planning involving consumables transfer.

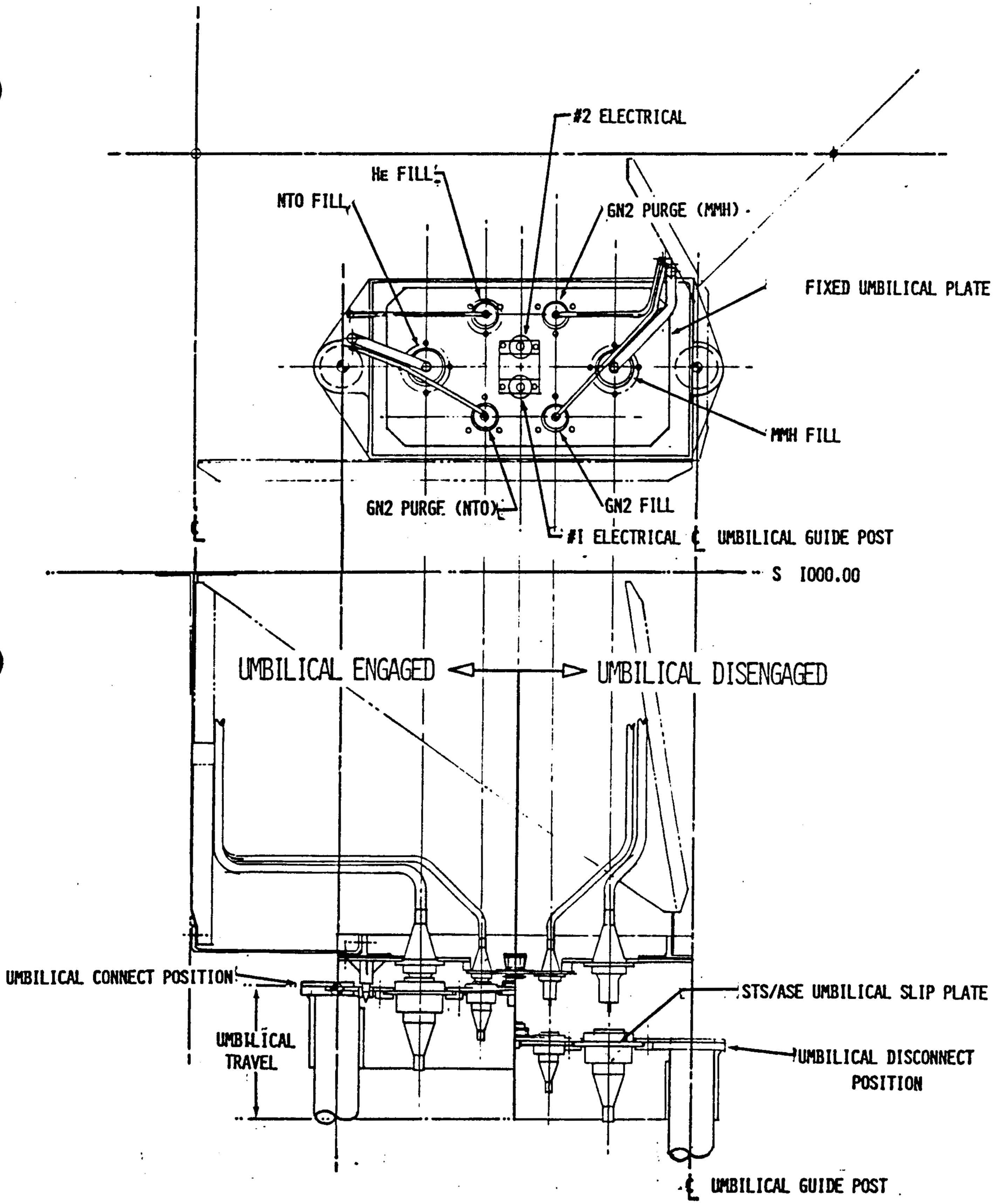


Figure 6.3-5 - "Gang" Umbilical General Arrangement

6.4 Optimization of Fluid Storage Capability

For consumables resupply using the OSCRS, the capacity of a GRO sized tank for different liquid consumables are 1240 lbs for hydrazine, 1070 lbs for monomethyl hydrazine (MMH), 1780 for nitrogen tetroxide (NTO), and 1150 lbs of water. For gas transfer, assuming a set of six bottles with 12.5" O.D., 24" in length, 20 lbs and 140 lbs of gaseous helium and nitrogen can be transferred per set.

Based on an annual projected consumable requirement, 3500 lbs of hydrazine, 3500 lbs of bipropellant, 6040 lbs of water, 120 lbs of gaseous helium and 840 lbs of gaseous nitrogen, required quantities of tanks of liquid consumables are: 3 tanks of hydrazine, 4 tanks of bipropellant and 6 tanks of water. For gaseous consumables, 36 bottles are required for each of nitrogen and helium.

The resupply schedule is evaluated assuming four resupply flights per year. On Flight Number 1, three tanks of hydrazine and 18 bottles of helium will be transferred which are 3780 lbs of consumables, total. Three tanks of water with 18 bottles of nitrogen will be supplied on Flights 2 and 4, resulting in 3870 lbs of consumable resupply for each flight. Finally, four tanks of bipropellant and 18 bottles of helium will be transferred on Flight Number 3, carrying 5760 lbs of consumables. Therefore, a total of 17,280 lbs of consumables will be transferred by the OSCRS per year.

At least two tankers will be required. The first tanker to be built will be with a hydrazine subsystem. The second tanker is envisioned to contain a water subsystem and an automatic remote interface. The first tanker will be updated for use at the Space Station with the automatic, remote umbilical interface.

Projected consumable transfer requirement is tabulated in Table 6.4-1. For the GRO sized tank, the capacity of hydrazine, MMH, NTO and water are tabulated along with the capacity of pressurant bottles in Table 6.4-2. For hydrazine and bipropellant, tanks are filled up to 93 percent, thus requiring a 15.5" diameter ullage bottle for each tank. For water, the tank is filled up to 87 percent. For pressurant bottles, a cascade of 6 bottles with 12.5" O.D., 24" in length is used.

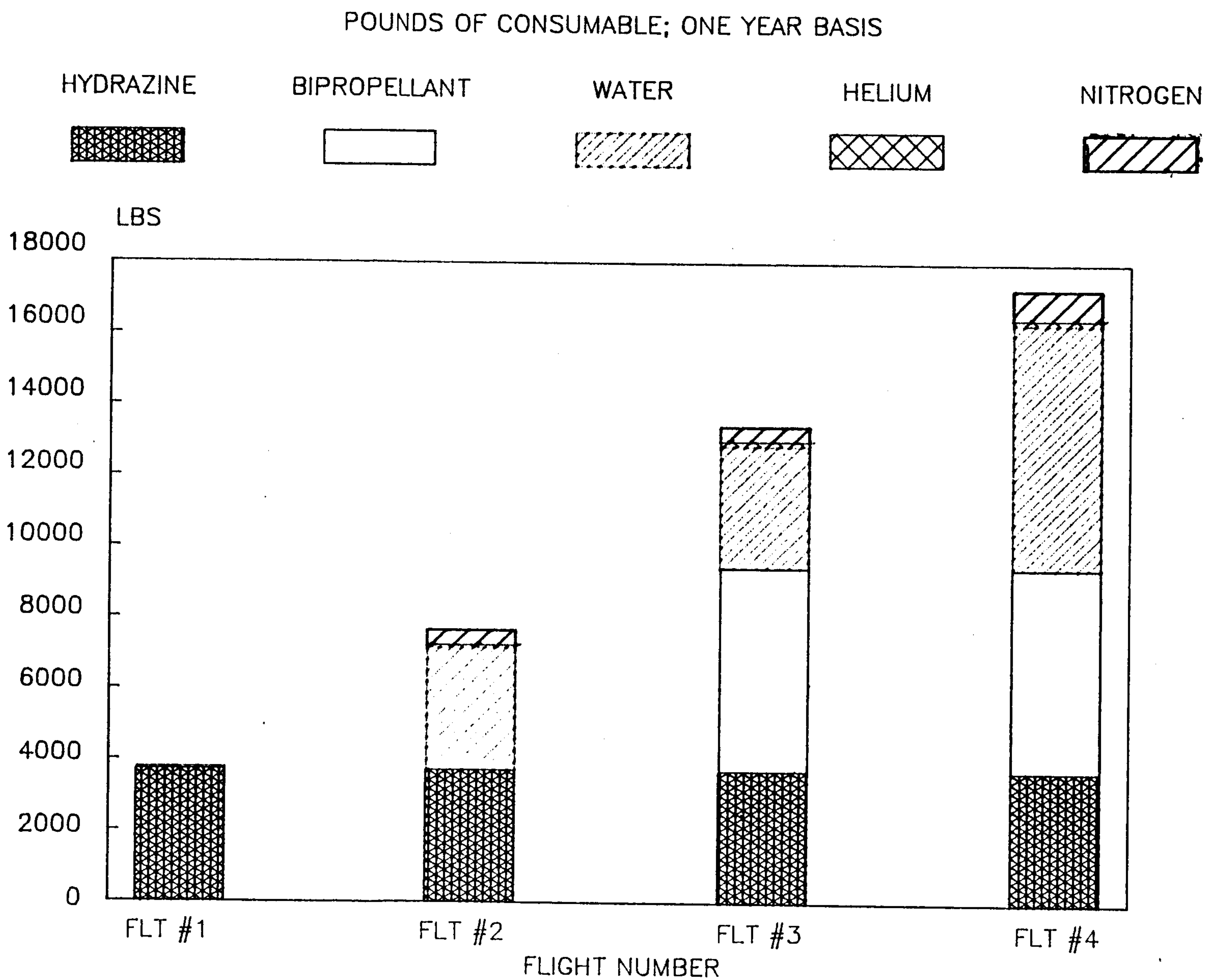


Figure 6.4-1 - Projected Consumable Resupply (Accumlative)

TABLE 6.4-1. REQUIRED CONSUMABLE TRANSFER

(AVERAGE ONE YEAR BASIC)

		WEIGHT (LBS.)	COMMENTS
LIQUID	N2H4	3500	2000 LB TO SPACECRAFT 1500 LB TO OMV
	BIPROP.	3500	TO SPACECRAFT
	WATER	6040	TO SPACE STATION
GAS	GHe	120	@ 4500 PSI, 70 F TO BIPROP. SYSTEM
	GN2	840	@ 3200 PSI, 70 F TO OMV

TABLE 6.4-2. TANK/PRESSURANT BOTTLE CAPACITY

GRO SIZED PROPELLANT TANKS @ 70 F	N2H4	1240 LB, 93% FULL
	MMH	1070 LB, 93% FULL
	NTO	1780 LB, 93% FULL
PRESSURANT BOTTLE* (PER 6 BOTTLES)	WATER	1150 LB, 87% FULL
	GHe	20 LB @ 4000 PSI 70 F, EOL
	GN2	140 LB @ 4000 PSI 70 F, EOL

*PRESSURANT BOTTLE WITH 12.5" O.D., 24" IN LENGTH
(8000 PSI CASCADE OF 6 BOTTLES)

In Table 6.4-3, the recommended consumable transfer schedule is tabulated assuming four resupply flights per year. It was attempted to make an even spread of each consumable transfer throughout the year. Also, the flights for water transfer are separated from the propellant transfer to prevent contamination of the potable water.

TABLE 6.4-3. RECOMMENDED CONSUMABLE TRANSFER SCHEDULE
QUANTITY TRANSFER: ONE YEAR BASIS

	NUMBER OF LIQUID TANKS TO TRANSFER			ULLAGE BOTTLES	NUMBER OF PRESSURANT BOTTLES TO TRANSFER	
	N2H4	BIPROP	WATER		GHe	GN2
FLT #1	3	0	0	3 - 15.5" SPHERICAL BOTTLES	3 X 6 = 18	0
FLT #2	0	0	3		0	3 X 6 = 18
FLT #3	0	4	0	4 - 15.5" SPHERICAL BOTTLES	3 X 6 = 18	0
FLT #4	0	0	3	--	0	3 X 6 = 18
TOTAL	3	4	6	--	6X6=36	6X6=36

Figure 6.4-1 presents the projected accumulative resupply totals on a one year basis.

6.5 Offloading Tanker to Tanker

The OSCRS tankers are expected to carry monopropellants (water and hydrazine), bipropellants (monomethylhydrazine - MMH and nitrogen tetroxide - NTO), gaseous nitrogen - GN_2 and gaseous helium - GHe. One application of the tanker is to remain on-orbit at the Space Station to resupply both the Space Station and spacecraft. Potential weight and cost savings can be realized if the on-orbit tanker (tanker to be returned) can offload residual propellants and pressurants to the replacement tanker. This study examined when the residual offloading will be justifiable.

Offloading pressurant (GHe or GN_2) from one tanker to another tanker was not found to be justifiable in the expected residual (100 lb or less) weight range. The additional structure, transfer interface, and required compressor would outweigh the quantity of pressurant to be transferred.

Figure 6.5-1 conceptually shows the required changes on the receiver side of the OSCRS tanker so that it can offload residual propellant from tanker to replacement tanker. The tanker will have a supply side that will remain unchanged from the present design. The supply side contains the standard end effector (SEE) for docking and the supply half of the quick disconnects, QDs. The receiver side will consist of a grapple fixture; and as depicted, carry either a monopropellant pair of QDs or a bipropellant set of QDs. The monopropellant interface requires only one QD for propellant transfer and one QD for redundancy. Ullage return QDs are not required since the Rockwell propellant transfer system concept is a blowdown system as opposed to a pressure regulated system which would require the ullage return QDs. The QDs will replace the fill and drain valves on the current design. A support and contamination plate (with required support structure) will also be added. The grapple fixture exists on the present design for attachment to an OMV. An avionic interface is also required. Tables 6.5-1 and 6.5-2 present weight and cost estimates of the added and replaced components required to offload the residual propellant. The cost estimates are recurring costs only since it is assumed that the components will be developed and exist for use on receiver spacecraft. Tanker weight increase to allow residual offloading is about 141 lbs for a monopropellant transfer and 155 lbs for a bipropellant transfer. Increased component costs are $623,000 for a monopropellant transfer and $891,000 for a bipropellant transfer.

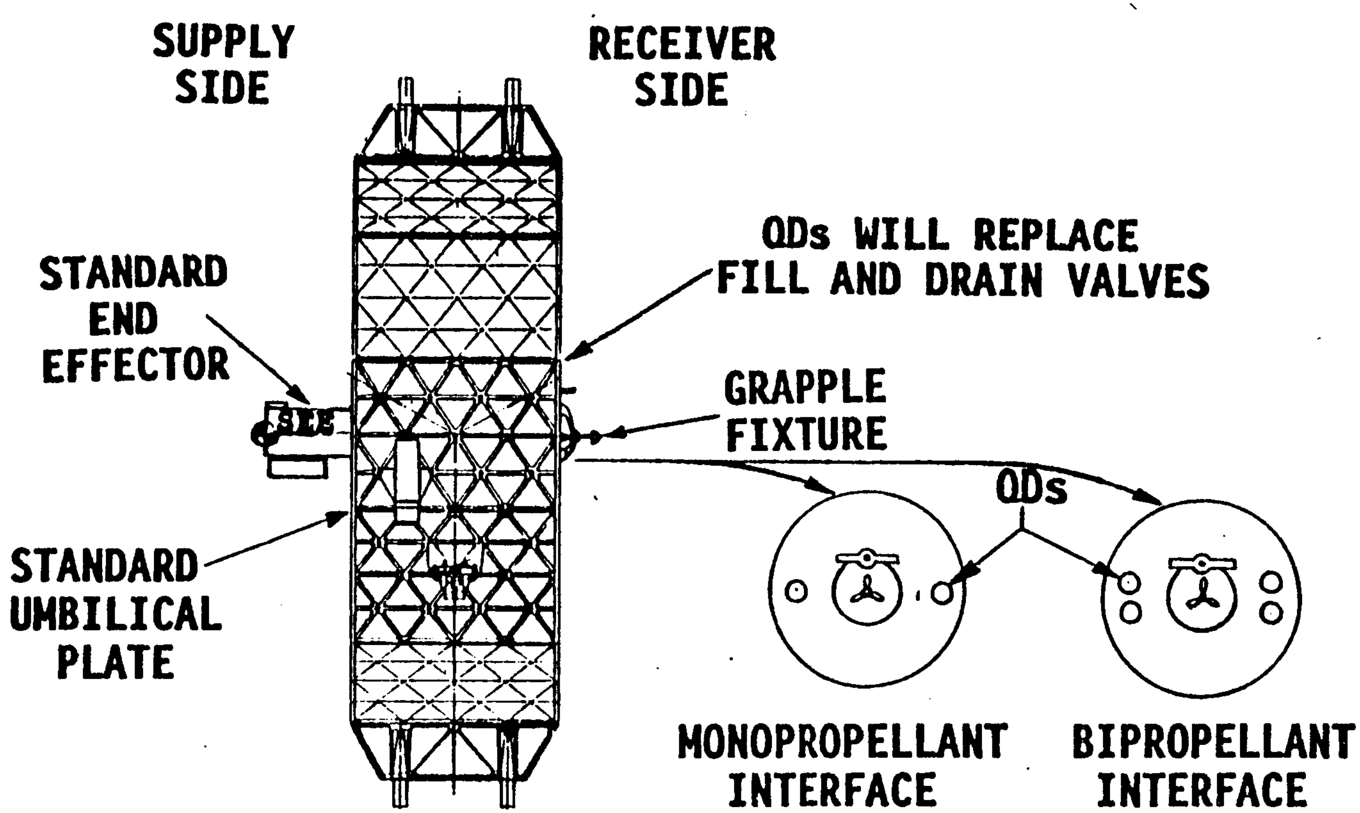


Figure 6.5-1 - Propellant Residual Offloading Interface Concept

Table 6.5-1 - Weight Estimate for Offloading Concept (lbs)

		each	monoprop	biprop
Added Components	support plate	12	12	12
	QDs and valves	8	16	32
	support structure	100	100	100
	avionic interface	15	15	15
removed components	fill and drain	2	2	4
Total:			141 lbs	155 lbs

Table 6.5-2 - Cost Estimate for Offloading Concept ($ K)

		each	monopropellant	bipropellant
Added Components	support plate	100	100	100
	QDs and valves	134	268	536
	support structure	125	125	125
	avionic interface	130	130	130
removed components	fill and drain	*	*	*
Total:			$623 K	$891 K

* Component removal will not result in a cost savings.

The results of the weight analysis indicate that a tradeoff of 141 lbs for the monopropellant system (versus 141 lbs or greater of transferred propellant) will be the basis for adding the receiver module to the tanker. But, cost is also a consideration. Launch costs can vary from 3,075 $/lb. (total launch cost based on 200 million per flight) to 865 $/lb. (using the weight load factor for STS transportation). The 3,075 $/lb was used in making the final decision, since this is what NASA must pay.

If the component cost is to be recovered on the first flight, then the component cost must equal the launch cost/lb times the difference of the residual offloaded propellant weight and the component weight. Or in equation form,

component cost = launch cost/lb (residual weight - component weight)

The trade-off point is where the component cost equals the total launch delta weight. A value of 344 lbs was calculated. This indicates that 344 lbs or greater of monopropellant to be offloaded in the first flight will pay for the nonrecurring component cost plus the cost to launch these components. Further flights will require the transfer of 141 lbs or greater of the monopropellant residual. The 344 lbs represents less than 30% of a 93% full GRO hydrazine tank. This is a reasonable quantity to justify the development of a resupply module subsystem. It should be noted that operational costs (i.e. to remove or add the resupply module) were not included in the cost analysis.

A value of 445 lbs or greater of the bipropellant to be offloaded in the first flight will pay for the nonrecurring component cost plus the cost to launch the components. The 445 lbs represents about 16% of the bipropellant in two (one MMH, one NTO) 93%, full GRO sized tanks. Further flights will require the transfer of 155 lbs to justify the addition of the resupply module.

The conclusions and recommendations from residual offloading are as follows:

1) Offloading pressurant from one OSCRS tanker to a second OSCRS tanker was not found to be weight or cost effective.

2) Offloading monopropellants can be justified by transferring at least 350 lbs of the monopropellant on the first flight (includes cost of components and full launch cost) and about 140 lbs for each subsequent flight.

3) Offloading bipropellants can be justified by transferring at least 445 lbs of the bipropellants on the first flight and about 155 lbs for each subsequent flight.

6.6 Thermal Effects of Station Basing

6.6.1 Thermal Control Attached to an OMV

The tanker model used in the monopropellant OSCRS analysis has been modified to provide a quick estimate of OSCRS performance when attached to OMV. Important changes to the math model are:

- All radiation interchange with the orbiter is removed and replaced with direct radiation to space.
- Average environmental heating rates for a $\beta = 0^{\circ}$, 270 nmi orbit are estimated and input to exposed surfaces. When facing away from the sun, the average is about 21 Btu/hr-ft^2.
- Surfaces facing the OMV have an averaged $\mathcal{F} = 0.05$ to space and experience no environmental heating.
- Effective forward face solar absorptivity(α) is increased to 0.5.
- Conduction to orbiter structure is deleted.
- The radiator is changed from a flat plate to a louvered design, set to cycle between 75°F (fully closed) and 105°F (fully open).
- During OSCRS sun facing attitudes, the average energy absorbed by the forward OSCRS face is 154.2 Btu/hr-ft^2.

The flight profile is a seven-day mission at space station altitude to refuel a satellite and return. OMV capabilities, provided by TRW, for a worst-case orbit, are two 20-hour "active" attitudes separated by 90 hours dedicated to OMV battery charging, for a seven-day "subsatellite" mission. The mission is illustrated in Figure 6.6.1-1. (If the angle is greater than zero or altitude is increased so that more charging time out of eclipse is available, deviations are possible.) In this analysis, the active attitudes are + x solar inertial attitude holds (OSCRS facing the sun). Propellant tanks are initially full, and are reduced to residuals at the half-way point in the mission.

As long as the OMV aft end faces the sun, levels of environmental heating on OSCRS are low, and results are expected to be similar for other β angles.

The analysis is conservative in that some of the heater locations in the model are located along insulation surfaces, resulting in increased heat loss to space. Effects of changes in OSCRS to interface with OMV and the satellite being serviced were not analyzed.

The OSCRS avionics system is estimated to require 140 watts of continuous power during the powered-down mode, in order to provide instrumentation data to the OMV and to operate the heater power supply RPC's.

The model indicates that 19 Kwh are required for the nominal mission, reflecting a possible slight hot bias in the initial conditions. A total of 24 Kwh, twice the energy required for the second (depleted) half of the mission is a more conservative estimate. These results scale up to 21 Kwh to 27 Kwh for the heater power requirement.

Total electrical energy including avionics power is about 51 Kwh for the worst case.

The use of $\alpha = 0.5$ results in a maximum surface temperature of 197°F, determined by a short analysis run and peak heat flux. All interior surfaces remain well below the propellant limits. Because large values of α can create problems during payload bay operations, $\alpha > 0.5$ was not considered.

To achieve the 0.5 forward face emissivity, about one-third of the front surface can be darkened to $\alpha = 0.8$. Orbiter OSCRS operations with the bay toward the sun (+ZSI) will be somewhat curtailed. Nevertheless, some sort of pre-deployment hot soak is desirable. The propellant and structure of the OSCRS can account for several Kwh. For example, with a full load of bipropellant, the heat capacity is about 1 Kwh per degree F for the propellant alone.

The avionics radiator louvers always open slightly, regardless of the temperature range selected, as long as the range is in a reasonable part of the avionics operational envelope. Heat losses at the radiator are significant: between 66 and 112 watts are lost, depending on the internal OSCRS conditions. With an 89 watt average loss, a total of 15 Kwh is radiated to space at the radiator surface. This is an area where substantial improvement may be possible.

The addition of the louvers and increase in radiator size required to tolerate sun exposure with the louvers results in a 16 lb weight penalty over the baseline OSCRS.

In order to more effectively utilize avionics waste heat, potential methods are:

- Use of heat pipe radiators as on the OMV;

- Use of internal isothermalization heat pipes to carry avionics waste heat to the OSCRS structure more effectively;

- Redistribution of the avionics boxes to allow more radiation and conduction to the OSCRS interior. this is a system-level change;

- Narrow the thermostat control range to reduce the maximum interior temperature of OSCRS. This would result in a reduced thermostat acceptance rate and increased cost but would also help the heater energy requirement.

These approaches will result in additional costs and weight, and should be traded against each other during the next phase of the program.

To control losses of heat, additional insulation can be added. However, beyond the existing 10 layers of MLI, the advantage of additional layers is difficult to quantify. The amount of heat passing through the MLI is reduced, but at this level some fraction of the losses are attributed to the configuration of the MLI blankets. Research into the most effective way of applying the MLI to the OSCRS might be worthwhile.

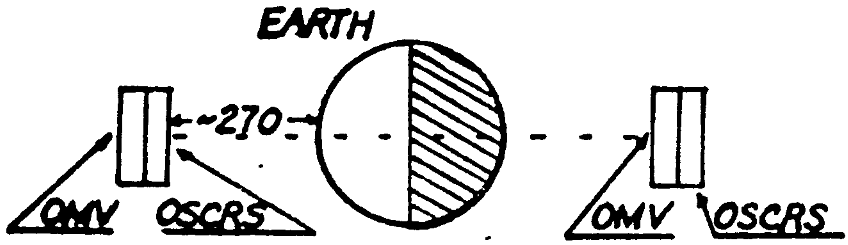
6.6.2 Thermal Control at Space Station

The objective of this analysis was to determine thermal and power parameters for OSCRS operation during space station basing. The space station is maintained in a constant local vertical attitude. The OSCRS maintains a fixed attitude relative to the earth and station during each orbit. Earth infrared heating on each surface is constant. Albedo and solar heating cycle with each orbit, depending on the orientation of OSCRS with respect to the station. The β angle (90 degrees minus the angle between the normal to the orbit plane and the earth-sun line) also influences solar orientation and eclipse duration.

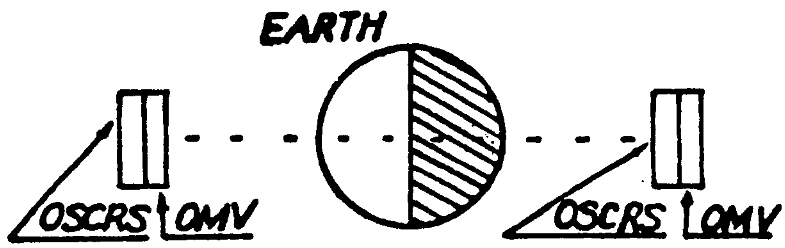
The earth-oriented station attitude results in selection of a continuous edge-to-earth OSCRS orientation. The radiator is continuously oriented at an angle of 30 degrees from the geocenter vector. Radiator solar exposure is limited by attitude and earth eclipse, but earth heating rates are fairly high. A 52-degree angle, the station maximum, was used to try to generate a moderately cold condition (Figure 6.6.2-1). During the available time, investigation to determine the worst cold case is not possible or worthwhile. The hot cases appear to be relatively benign, and not much different from the cold cases. Both hot and cold cases can be dependent on station configuration. Shadowing and reflections onto the OSCRS by the station are not considered.

The OSCRS is assumed to be configured as it is for the OMV mission. The four tank consumables resupply tanker model was used. The forward surface absorptivity is 0.5, and radiator louvers are used. Two cases are analyzed. In the first case avionics power is 380 watts, representing a full operational status similar to that required during the GRO refueling mission previously studied. In the second, avionics power is 140 watts, a powered-down state typical of an OMV coast phase. Avionics power and redundancy levels required for space station operations have not been defined, thus both cases were analyzed. Conduction to the environment, through the keel fitting only, is restored to the OSCRS model to simulate OSCRS conduction to station structure.

Figure 6.6.1-1 - OSCRS/OMV Mission Model

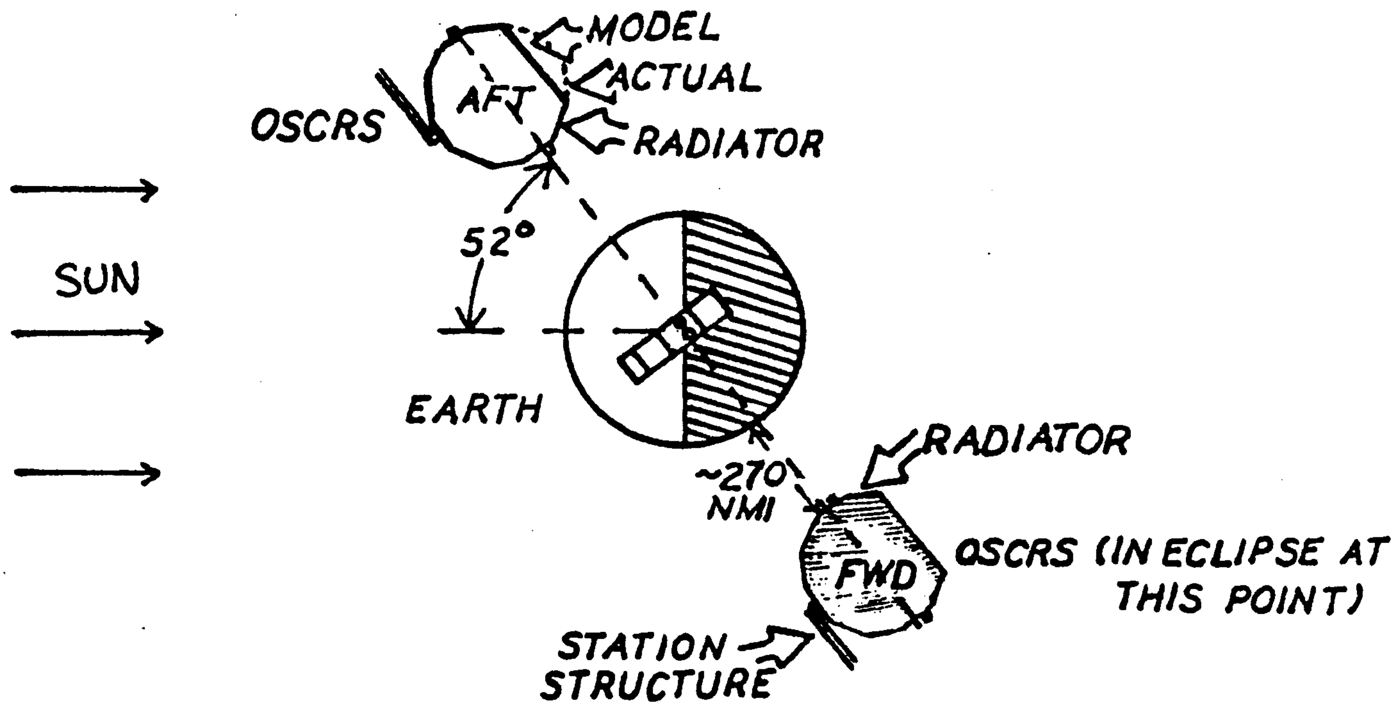


Attitude 1 Required by OMV



Attitude 2 Held Twice, 20 hr. Each, With 90 hr Gap in Between

Figure 6.6.2-1 - OSCRS Orientation Showing Fixed Attitude With Respect to Earth



Environmental heating on the OSCRS was based on average values available from a prior flat plate orbital heating study. The data was factored down to account for a 270 nmi station altitude, vs. 150 nmi used in the study. All OSCRS surfaces except the radiator and forward face were evaluated at a solar absorptivity (α) of 0.30. The radiator is hot-biased slightly using $\alpha = 0.27$.

An attempt was made to cold bias the subsystem temperatures slightly at the start of the analysis run, to avoid generation of optimistic power levels. This approach was successful except in the area of the propellant, which proved to be slightly hot biased in some of the tanks.

A separate analysis was performed manually to determine the capability of the louvered radiator surface to tolerate solar exposure. The radiator area utilized in the TMM is limited to about 13.4 ft^2. An actual 14.6 ft^2 to 14.7 ft^2 is actually available on the avionics bay outer surface. The radiator must remain below about 135°F to 140°F in order to maintain the avionics below its 158°F limit. The interior area of the radiator is assumed to be effectively reduced by the avionics boxes. For this analysis, outer panel area is 14.6 ft^2 and is a maximum of 0.23 at 30 degrees from perpendicular sun incidence.

The results of the cold case analysis are shown in Table 6.6.2-1. Heater power results presented in the table are scaled up slightly from those generated by the math model to reflect OSCRS dimensions. The results show that in the worst case, OSCRS utilizes 79 kwh per week at 380 watts avionics power, and 44 kwh per week at 140 watts avionics power. Average worst case power loads are 0.47 kw and 0.26 kw, respectively.

Heater energy requirements are less during the first (fully loaded) half of each mission than during the second (depleted) half. This is partly due to the slight hot bias of the propellant tanks, which delays onset of the first heater cycle. The thermal control system is also more efficient when the tanks are fully loaded because the cooldown portion of the heater cycle is increased. The system may then reject heat at a slightly lower average temperature. The system warmup is rapid and the relatively higher average temperature, compared to that which occurs when the tanks are depleted, has little influence.

Because of space station orientation, the solar heating rate on the radiator constantly changes. Because of the louvers, the effective absorptivity changes with sun angle. A maximum solar heating rate of about 88 Btu/hr-ft^2 can occur. A total environmental heating rate of about 134 Btu/hr ft^2 can occur at sunrise under the worst conditions.

At 140 watts avionics power, the avionics radiator can tolerate at least 142 Btu/hr-ft^2. (If the internal blockage by the avionics is favorable, 145 to 172 Btu/hr-ft^2 could be tolerated).

At 380 watts, the worst case design can tolerate at least 86 to 92 Btu/hr-ft^2, depending on radiator temperature, with a potential range of 89 to 116 Btu/hr-ft^2.

During transfer of multiple fluids, higher power levels could be experienced briefly. At 540 watts, the worst case produces a maximum tolerable environmental heating rate of 48 to 55 Btu/hr-ft^2.

Prolonged sun viewing by the radiator with near perpendicular sun incidence angles is unlikely during space station operations because the station attitude is earth-tracking and the sun angle is constantly changing. A non-louvered radiator (OSCRS orbiter operations baseline) has a lower effective absorptivity and can be designed to tolerate continuous solar exposure within the available radiator space.

It is concluded that the OSCRS design, as modified for OMV operations, will work in a space station environment. Average heater power is less than 122 watts as long as the avionics system is operating.

TABLE 6.6.2-1. THERMAL ANALYSIS RESULTS

Heater Power Scaled up 12% for OSCRS
Dimensions and Geometry
One Week Mission, Start time = 16 hours, End Time - 184 hours

	AVIONICS 380 WATTS	POWER 140 WATTS
TOTAL HEATER ENERGY 168 HRS (KWH)	12.1	18.2
HEATER ENERGY, FIRST HALF WEEK (KWH)	4.6	7.9
HEATER ENERGY, SECOND HALF WEEK, DEPLETED TANKS (KWH)	7.5	10.3
TOTAL SYSTEM ENERGY HEATER + AVIONICS (KWH)	76.0	41.7
TOTAL SYSTEM ENERGY FIRST HALF WEEK (KWH)	36.5	19.7
TOTAL SYSTEM ENERGY SECOND HALF WEEK (KWH)	39.5	22.0
TOTAL SYSTEM ENERGY SECOND HALF WEEK TIMES 2 WORST CASE ENERGY UTILIZATION (KWH)	78.9	44.1
AVERAGE WORST CASE		
--TOTAL LOAD (KW)	0.47	0.26
--HEATERS (KW)	0.09	0.12

6.7 Station Basing of Tanker Elements

The basic OSCRS tanker philosophy maximizes commonality between the basic monopropellant tanker required to resupply GRO with 2500 lbm of N_2H_4 and future earth storable fluid resupply tankers of over 8700 lbm of fluids. this commonality was optimized by use of the hybrid tanker concept which has a common structure for all earth storable propellant tankers (with less than a 70 pound penality to carry the full fluid load compared to the 2500 lbm load); and modularizes all subsystems so that only mission essential elements be flown in any mission scenarios.

The modules are designed to replace existing modules (e.g. a water subsystem module for a hydrazine subsystem module) or to be placed in predetermined structural regions. The structure is machined open grid aluminum alloy capable of holding six GRO size propellant tanks, and contains growth space for pressurant tanks as well as space for the control avionics to support any unspecified mission requirements. All module replacement was designed for ground operations.

The basing of OSCRS tanker elements was determined to be limited to required shielding against micrometeorites and other space debris. Tanker protection for long duration on-orbit storage needs to be provided. A pea-sized object colliding with a spacecraft at 6 miles per second can have the destructive power of a hand grenade, and there are estimated to be over 50,000 fragmentary objects larger than a pea in orbit. Even a much smaller object can cause significant damage (eg. in 1984 the Challenger returned from a mission with a one centimeter wide pit in a windshield pane from a paint flake only 0.2 millimeters wide). Current Space Station design is to add shielding to protect from impact of one centimeter size projectiles; OSCRS will require the same protection.

A second potential area of Station basing of tanker elements could be a power module or individual battery packs to be added to OSCRS for any unspecified mission requirements. Figure 6.16-7 indicates potential battery locations on the tanker.

6.8 Shuttle-to-Station Transfer of OSCRS Hardware

In transferring the OSCRS from the Orbiter payload bay to the Space Station for basing at the Station, it is desirable to maintain power to the heaters to prevent potential freezing of fluids. Depending on the fluids being carried, either freezing or thawing may cause damage to the fluid components. The development of the potential problem depends on the timelines required to transfer and the orientation of OSCRS to the thermal environment. The timelines are subject to cable connector removal and replacement times. Under reasonable assumptions the RMS transfer of OSCRS to the Station will take place without heating and instrumentation being required; but any problem in maintaining timelines indicates the necessity for both heater power and instrumentation to maintain and verify safe conditions during the transfer.

OSCRS is in a quiescent state during movement from one location to another. The recommended system does not have two fault tolerance to provide instrumentation during a move. One failure tolerance can be obtained by adding a second umbilical to the SEE.

In a typical sequence, the SEE umbilicals would be connected to the OSCRS; then the Orbiter connectors would be removed; physical transfer would take place; the Space Station connectors would be connected and the SEE would be removed. Once the SEE is connected, the Station would be receiving data for monitoring the OSCRS as well as furnishing power to the heaters. Before the SEE umbilical is removed, the permanent Space Station connectors would be in place. This suggested sequence would allow one failure to occur, while maintaining power and data flow to and from the OSCRS.

6.9 Central Versus Multi Location Refueling Options at Space Station

Transferring the OSCRS from one location to a second location at the Space Station will create a similar situation to the Shuttle-to-Station transfer discussed in section 6.9. It was recommended that OSCRS is to remain "hooked-up" at all times; that one set of umbilicals be connected before the second set of umbilicals are removed. This approach will allow one failure to occur, while maintaining power and data flow to and from the OSCRS.

6.10 On-Orbit Operations at the Space Station

Prior work on the OSCRS has been related to operation in the payload bay of the Space Shuttle. Operation at and attached to the Space Station necessitates a re-evaluation of user requirements and the effects of those requirements on system operation, interfaces and software.

The requirements of OSCRS at the Space Station include the following:

- One failure tolerant for mission completion
- Two failure tolerant for safe operation
- Operation in either the payload bay or at Space Station

The principal new requirement for operation at the Space Station is that of 'dual mode' operation. It must be possible to transport and operate the OSCRS in the Orbiter payload bay and to transfer the OSCRS to the Space Station and subsequently operate it there. During the transfer, both the Orbiter and the Space Station will be connected to the OSCRS.

Several interface approaches were considered. A simple replication of the AFD equipment in the Orbiter would require little redesign, however many wires carrying special functions for override and emergency operation are used. After consideration, it was concluded that a different approach to providing failure tolerance to the second failure and for emergency operation was appropriate.

The control panel and computation system selected for the revised OSCRS (Figure 6.10-1) which will provide the capability to operate from the payload bay of the Orbiter and the Space Station is a four string system. In the Orbiter, four control panel sections similar to the three AFD panel sections originally recommended for operation in the payload bay provide control and display of the commanded function or functions. Continued use of the Grid computer or similar graphic display on the AFD will provide flexible displays for observation of multiple system functions. Caution and warning will remain as a separate tie to the Orbiter C&W system.

At the Space Station, the same interfaces will provide control and display of commanded sequence(s). Both function control and display and flexible system detailed displays could be provided on a standard Space Station 'Multipurpose Application Console'.

The approach chosen will provide operation in the payload bay and from the Station while offering single failure tolerance for completion of operations and two failure tolerance for safety without the necessity for operator intervention. While the operator may intervene, the system is capable of isolating the first failure to a string by string comparison and powering down the failed string so that if a second failure should occur, safety will not be jeopardized. The operator will be notified of the second failure, at which time fuel transfer operations should be terminated and the system safed. While the system is still capable of fuel transfer operations, it is no longer failure tolerant and operations should be continued only for reasons strong enough to warrant the serious safety risks involved.

6.11 Use of OSCRS with OMV

The OSCRS must be able to transfer fuel while attached to the OMV as an OMV payload. This capability must extend to operation at or near the Space Station, near the Orbiter or at a remote location. In addition, the OSCRS must operate in a safe quiescent mode (no fuel transfer) while on an ELV during prelaunch, launch and on orbit. It must also operate safely during transport in the Orbiter payload bay and while stored at the Space Station on or off the OMV between missions. Previously, the OSCRS operated only in the payload bay. In the OSCRS design for operation in the Space Shuttle, numerous functions are directly wired from the aft flight deck to the OSCRS avionics. Included among these are the bank select power switching for fault isolation/fault effect avoidance, the safing circuits to override all other control in the event of massive failures, the control and sequence number feedback from the 'control panel', pyrotechnic actuation switches for emergency separation and power control for the FMDMs and heaters. For remote operation, these functions must be handled in a different way.

Requirements for failure tolerance in the remote operation need to be critically examined. Present OSCRS design is predicated on manned space flight system guidelines. Separate cables, connectors and boxes for each string are not routinely employed in unmanned satellites as is the practice for manned systems. Those portions of the remote configuration of OSCRS could be designed to satellite guidelines rather than manned flight guidelines providing suitable safety is provided in the payload bay and for operations in proximity to the space shuttle and the space station.

Failure tolerance is ultimately dependent on available power and due consideration must be given to the number and switchability of sources in establishing compliance or deviation to the stated system requirements.

OSCRS operation in the payload bay was accomplished by three string control from the Orbiter aft flight deck (AFD). With three active redundant strings, the system remained safe and operational after one failure. After the second failure, re-configuration through bank switching or a special safing operation was used to make the system safe. This re-configuration was accomplished through the use of separate circuits from the AFD of the orbiter.

To maintain the same type of backup while operating on the orbiting maneuvering vehicle (OMV), the separate circuits used in the Orbiter would have to be 'recreated' in some way to be independent of the normal three redundant strings.

Operation at the space station requires similar consideration. Although using separate circuits with the station is a possibility, use of numerous dedicated circuits opposes Station data management system access to the ongoing refueling process.

In order to operate OSCRS in any of the three environments (orbiter payload bay, space station or on OMV) it seemed prudent to attempt to arrive at a suitable common or similar interface between OSCRS and each of the three basing points. Several approaches were examined in a trade study performed to examine advantages and disadvantages of various methods.

A transliteration of the requirements into a mechanization involves three strings together with the parallel functions split out through serial inhibits in each string. Possible configurations were examined and found lacking. Instead, a four string system is envisioned. This approach reduces the number of wires required and gives a non varying technique for the control and monitoring of operation. It can be applied to operation on the OMV, at the Space Station and in the payload bay of the Orbiter.

A four string avionics system provides the required safety and control survivability and is amenable to operation not only on the OMV but also in the Orbiter payload bay and at the Space Station. Four strings using standard protocols can be adapted to 'neck up' or 'neck down' at communications interfaces and thereby use available OMV channels as long as the requisite safety and survivability are assured.

Communications security is related to the communications channel provider (OMV, Space Station, ELV, ground station, etc.). Should this prove undesirable for any reason then further consideration would be necessary.

6.11.1 Failure Tolerance with OMV

Based on the assumption that the OMV will provide two independent communications channels, then two failures can cause loss of control. This means:

A. During transit, the OSCRS will be quiescent and safe but only one failure tolerant to provide information (temperatures are of primary interest for safety).

B. During fuel transfer, the OSCRS will be active and may continue fuel transfer after one failure. Safety information would not be available following loss of the second communication channel. OSCRS can be configured to sense loss of both communication channels and automatically terminate fuel transfer and safe the system.

Loss of both power sources during fuel transfer may create a dangerous or terminal condition. Fuel transfer operations on OMV are not recommended in the proximity of either the Orbiter or Space Station when OSCRS is attached to the OMV.

6.12 OSCRS Design Impacts Associated with OMV and Space Station Operations

The requirements of OMV and Space Station operations has indicated a need for the development of several modules or design changes to the preliminary monopropellant OSCRS tanker. These changes include a water subsystem module, a remote/automatic umbilical interface, a residual offloading interface, thermal subsystem adjustments, increasing the avionics subsystem from a 3-string (man-in-the loop) to a 4-string (remote) system, and a potential power module addition.

The water subsystem module does not change the basic monopropellant OSCRS tanker design. The monopropellant OSCRS tanker was designed as a generic tanker where one fluid subsystem could be replaced by another fluid subsystem. A more detailed discussion of the water subsystem module can be found in section 6.2.

A remote/automatic umbilical interface was designed to be added to the existing monopropellant tanker concept. Calculated structural weight for the interface is about 170 lbs. The design combines all interface functions for spacecraft capture, rigidization, retrieval, umbilical engagement and contamination control in one unified, sequentially controlled operation. A more detailed discussion of the remote/automatic umbilical interface can be found in section 6.15.

FIGURE 6.10-1 FOUR-STRING SPACE STATION OSCRS

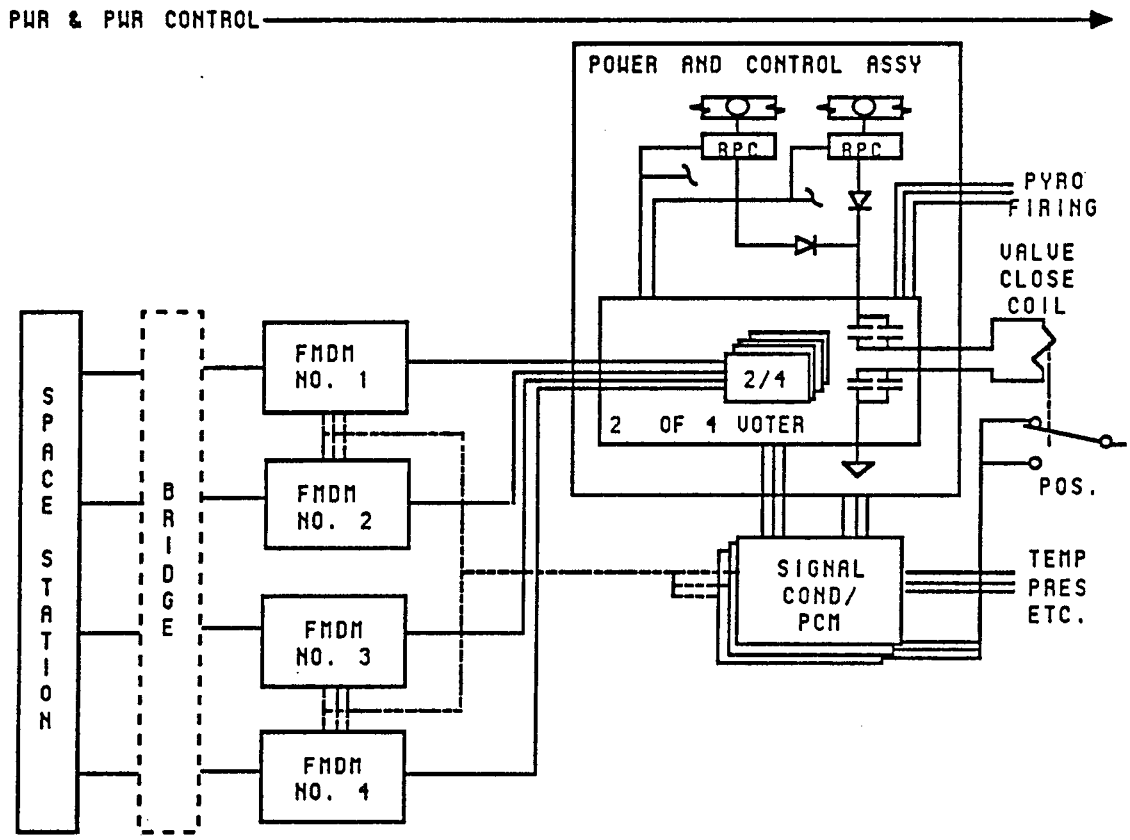
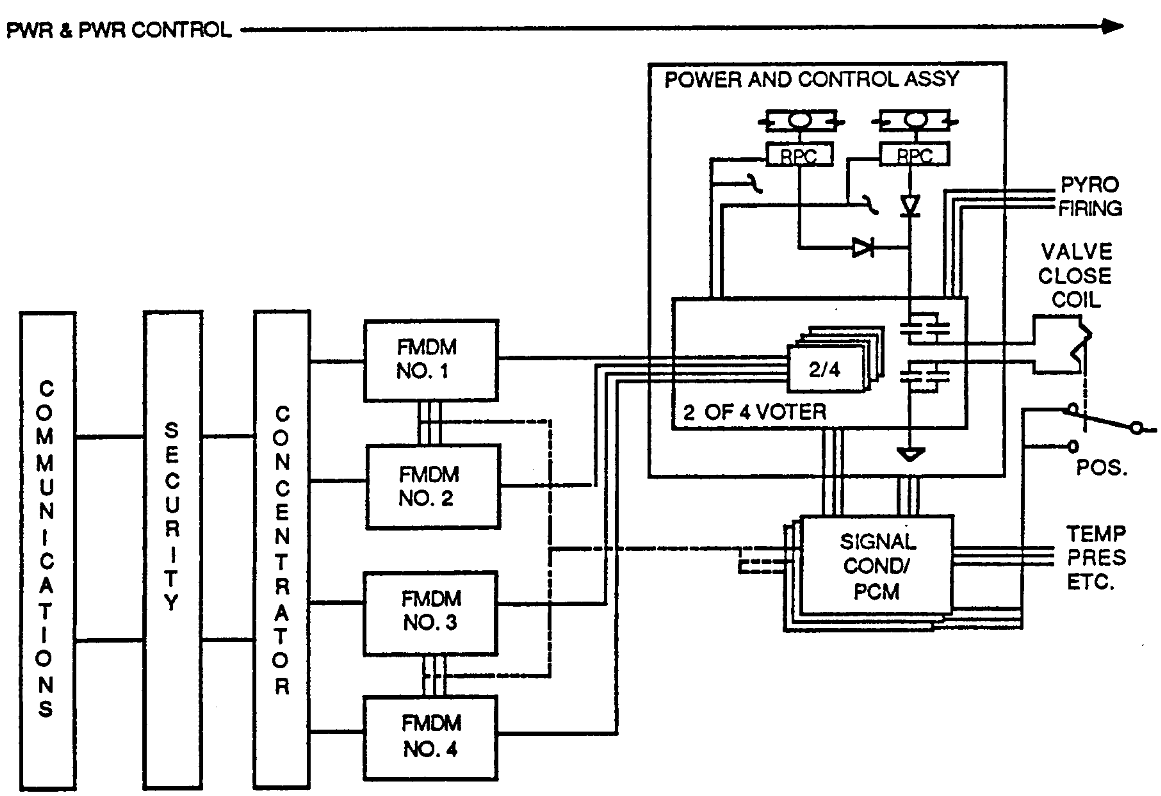


FIGURE 6.11-1 FOUR-STRING OMV OSCRS



Required changes on the receiver side of the OSCRS tanker so that it can offload propellant from tanker to replacement tanker is shown conceptually in Figure 6.5-1. The residual offloading module weights about 141 lbs. for a monopropellant transfer and 155 lbs. for a bipropellant transfer (Table 6.5-1).

Changes to the thermal subsystem for OMV operation includes increasing the solar absorptivity to 0.5, increasing the radiator size (the radiator removes avionic waste heat), and adding louvers. The addition of the louvers and the increase in radiator size required to tolerate sun exposure with the louvers results in a 16 lb weight penalty over the baseline OSCRS.

A one week mission will require about 51 Kwh, which the OMV cannot supply as presently funded. There are two areas of future study required; effective utilization of avionics waste heat (about 15 Kwh), and the addition of a power source on the OSCRS.

To more effectively utilize avionics waste heat, potential methods are:

- o Use of internal isothermalization heat pipes to carry avionics waste heat to the OSCRS structure more effectively;

- o Redistribute the avionics boxes to allow more radiation and conduction to the OSCRS interior.

Power can be added to OSCRS by adding batteries to the OSCRS structure. In addition, solar panels can also be added to the external perimeter if the battery requirement can be provided internal to the OSCRS structure.

Design of the current avionics system was predicted for operation in the payload bay and tolerance to one failure for mission completion and two failures for safety.

The OSCRS 'payload bay' avionics employs a system with three active strings. This system was designed to operate in the payload bay and to tolerate a single failure without either failure detection or intervention of any kind. It therefore has the capability to continue a mission to completion after a failure. It was further required and designed to allow safing after a second failure in order to assure safety of the Orbiter and crew after two failures.

Safing and emergency separation after the second failure are accomplished by means independent of the three active strings and really are in addition to the triply redundant system. Wires from switches on the OSCRS control panel in the aft flight deck (AFD) of the Orbiter are used to control these functions. This system provides not only emergency control after two failures but provides it through dissimilar control and is therefore impervious to 'generic' failures in the triply redundant strings.

The system has many desirable features and is a good choice for operation in the payload bay. The large number of wires from the AFD and the control circuits they represent would probably be excessive for interconnection to the Space Station and certainly for the orbital maneuvering vehicle (OMV). Power control should remain separate so that the supporting entity has veto authority over the OSCRS but the other functions are likely better integrated into the main control stream. Such integration is the object of several system alternatives for remote operation which are examined as candidates for use in operation at the Space Station or on the OMV.

Due to the inherent difficulties in determining which string has failed in a system having only two remaining strings operational, a four string system was examined. A four string system has the advantage that all strings may be validated by comparison to other strings. One string may be deactivated after the first failure and string outputs monitored thereafter as in a three string system. Only if three failures in the avionics system are to be tolerated is there the difficulty of isolating the third failure. The proposed four string avionics system is shown in Figure 6.11-1.

A four string system has the advantage of being easily configured to handle automatic faultdown through two failures in a consistent manner rather than using peculiar work-arounds. It has the disadvantage of being more susceptible to potential 'generic' faults since it uses four identical strings. It is amenable to an interface which requires fewer wires and fewer types of command structures than a system using different 'overlays' to achieve fault tolerance.

The small number of wires and ability to 'neck-up/neck-down' at a data bus interface makes the four string system more adaptable to use with the Space Station and the OMV.

6.13 End Item Specification (EIS)

The EIS was developed as the basis for the design, development, fabrication, certification, and operation use of the OSCRS. It was published as a separate report, STS-86-0272.

The unique requirements resulting from this study for a water tanker system, an automatic refueling interface design, and a remote operations avionics system have been incorporated as appendices to the basic EIS.

6.14 Monopropellant (Water) OSCRS Phase C/D Program Plan

The monopropellant (water) OSCRS Phase C/D program plan defines the scope and schedule of all development elements. The plan consists of a work breakdown structure (WBS) (Figure 6.14-1), supporting schedules (Figure 6.14-2), and identification of task interaction (Figure 6.14-3).

The complete detailed program plan is documented in DRD-8 report number STS 87-0268. Key features of the plan are summarized below.

The plan provides for a high-fidelity mock-up engineering aid to be built after the preliminary design review. The engineering aid which allows early hands-on design assessment will be available for the critical design review. The engineering aid will be used for crew and safety reviews, crew training, manufacturing aid, facility interface tool, and GSE/Handling design aid.

The program plan incorporates a make-or-buy-plan to use low cost flight proven hardware and designs, provide open competition for components unique to OSCRS, use existing facilities, and involvement of small and minority-owned businesses in the development/fabrication of OSCRS.

A detailed verification approach is defined in the program plan. It includes definition of verification requirements, verification plan for components, subsystems, systems, verification methods (analysis or test), and verification of flight operation functions with simulated vehicle interfaces and launch/space environment.

Definition of the fabrication approach for OSCRS is based on using the Payload Integration Nominal Cost Hardware (PINCH) management concept. This concept provides for a dedicated centralized collocated team with the build and flow plan under control of the program manager. The fabrication process will use simplified tooling and the engineering aid to minimize cost. Fabrication will be accomplished in phases: structure and panels, mock-up and assembly, integrated tests, refurbishment, acceptance test and delivery.

The plan also defines/implements safety and quality control elements which assure conformation to specified design and performance criteria.

There are two major differences between the water tanker program plan and the hydrazine tanker program plan. The first difference is the addition of the remote/automatic umbilical interface to the water tanker. The second difference between the program plans is that the hydrazine tanker will precede the development of the water tanker by one year and the full system testing will occur on the water tanker before usage of the hydrazine tanker.

6.15 Automatic Refueling Interface Design

On-orbit consumables resupply offers the spacecraft/satellite community unique opportunities for extending the useful life of their vehicles. The potential also exists for supplying spacecraft with their initial propellant load after launch. This permits lighter structures through reduced launch loads which increases payload capability.

Routine resupply of various consumables, including potable water, must become a common occurrance to meet the needs of the community. To support these resupply mission requirements, a NASA-Industry standard umbilical interface is required. The objective of this study was to define a baseline remote/automatic consumables resupply interface design concept as the NASA-industry standard.

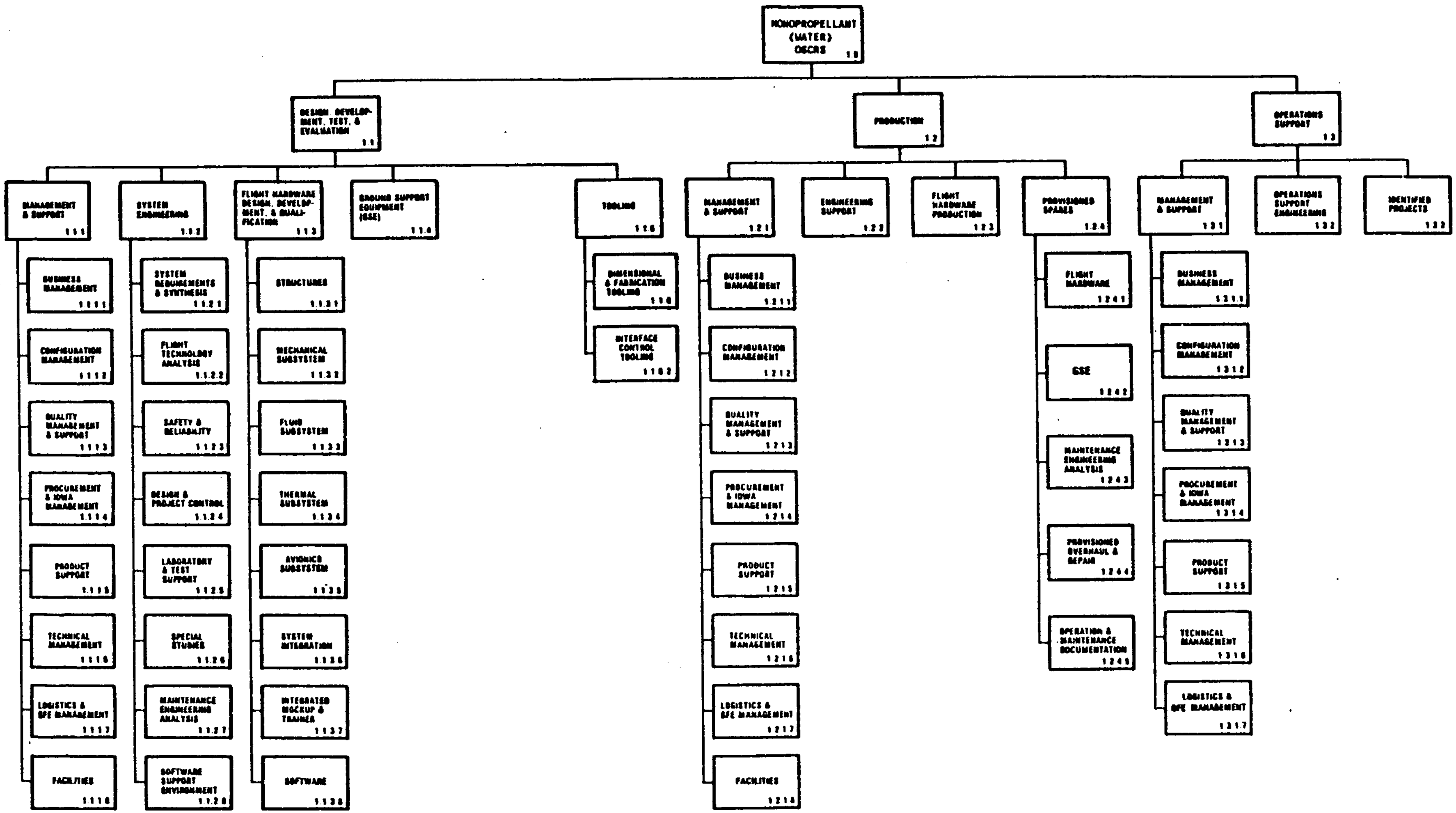


Figure 6.14-1 - OSCRS Monopropellant (Water) Tanker C/D WBS

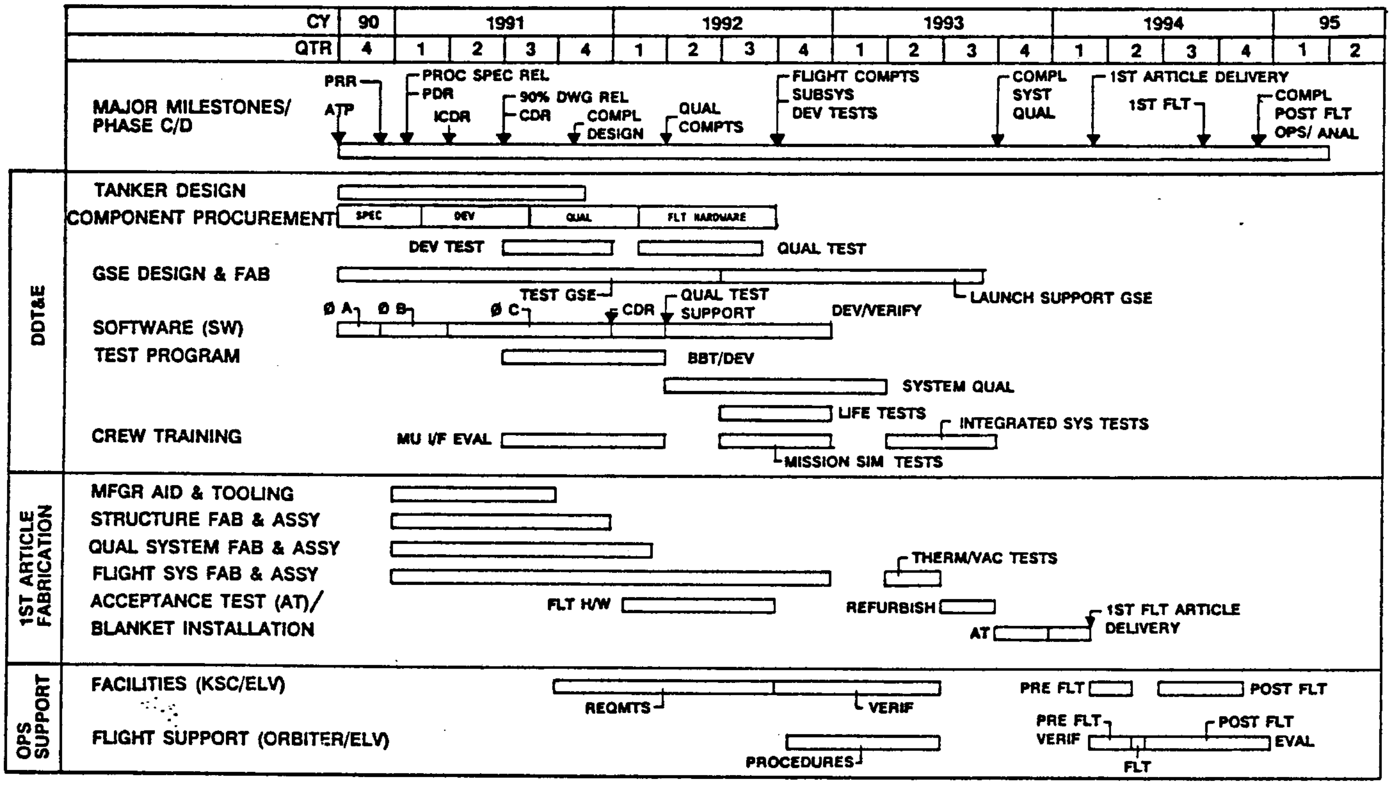


Figure 6.14-2 - OSCRS Monopropellant (Water) Tanker Phase C/D Program Schedule

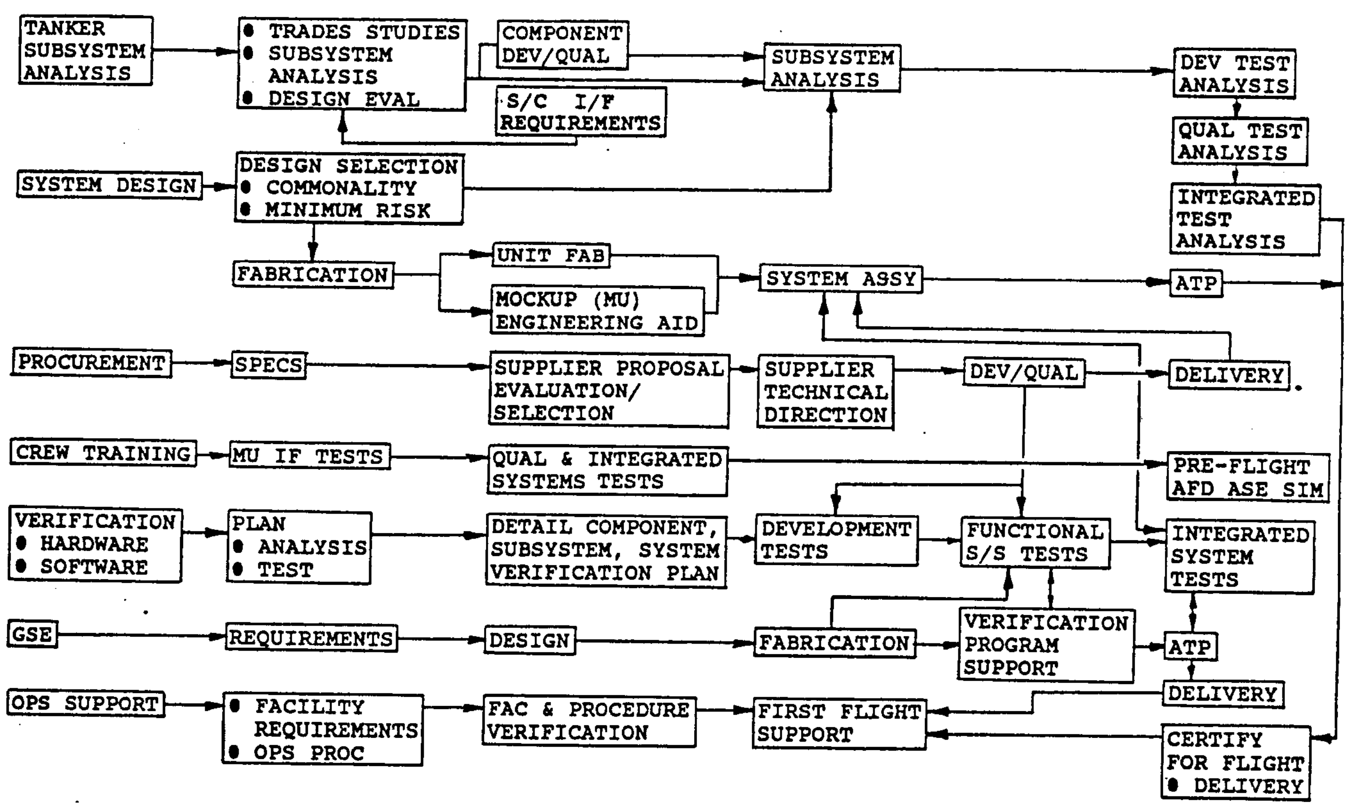


Figure 6.14-3 - Task Interaction Flow Diagram

Previous studies limited this effort to STS Orbiter payload bay operations. Present and foreseeable limitations in orbiter operations (cargo manifesting limitations) confined with expanding scope of spacecraft and Space Station resupply needs, requires a broader understanding of interface functions and requirements.

There are two sides to any interface. One side is integral to a resupply vehicle (OSCRS, OMV) and the other to a mating spacecraft. The interface concept resulting from this study relates to the supplier half of an OSCRS configuration. Although the berthing/umbilical support structure/interface is directly related to the Rockwell baseline OSCRS configuration, the actual berthing/umbilical interface must be adaptable to the selected resupply vehicle configuration with minimum perterbation. The interface should be designed such that the spacecraft half contains only those components required to complete the transfer of fluids.

6.15.1 SPACECRAFT CAPTURE & BERTHING

For OSCRS operations a distinction should be made between berthing and docking. For this study berthing between OSCRS and other spacecraft (including space station) is defined as using the controlled rate of a capture/retrieving mechanism mounted on OSCRS to assure a soft initial interface contact with little or no kinetic energy absorption associated with conventional docking speeds and masses. This type of interface control provides OSCRS with a wide adaptability to berth spacecraft of differing geometry and masses while protecting protruding or extended equipment on either spacecraft from adverse "G" loads.

The baseline OSCRS configuration incorporates two separate and different mating vehicle interfaces. Figure 6.15-1 illustrates this configuration. The OSCRS to orbiter interface utilizes the standard payload bay longeron/sill and keel trunnions. The OSCRS to spacecraft interface features three removable MMS/FSS latches. The baseline OSCRS design also includes a flight releasable grapple fixture (FRGF) to facilitate on-orbit relocation in the orbiter payload bay utilizing the standard end effector ("SEE") attached to the end of the remote manipulator system (RMS) arm.

One OSCRS operational requirement is to interface with the orbital manuvering vehicle (OMV). Figure 6.15-2 evaluates three berthing concepts. Two of the three propose mating a "SEE" to a redundantly driven extension/retraction mechanism providing a payload interface 27.5 inches in front of a vehicle facing plane. The third concept consists of two components; a probe and receptacle cylinder. The probe, mounted on the spacecraft, travels a total distance of 5.5 inches into the receptacle (OSCRS half), compared to the 27.5 inches of the other two translating "SEE's". This concept is not intended to deploy/retract and as such the limited linear travel is a disadvantage to a common berthing/umbilical design.

One "SEE" was lost in the Orbiter "Challenger" accident. Four remain and two will be modified to new load requirements by the SPAR Aerospace Ltd., RMS Division, Weston, Ontario, Canada presents the modified "SEE" load capabilities.

6.15.2 UMBILICAL PLATE DESIGN

In defining an umbilical interface, fluid, gas, and electrical/avionic connectors should be evaluated on the basis of reliability, operating characteristics, weight and cost. The numbers and sizes of umbilical connectors must be determined. To accomplish this, a consumables resupply "maximum" scenario must be developed. In this study the established maximum scenario consists of transfering biopropellants (MMH/NTO) utilizing ullage return. Additional transfer of gaseous helium and nitrogen, as well as connecting the electrical/avionics, is required. Redundancy for all connectors is assumed.

Table 6.15-1 presents connector count rationale and totals individual connectors. The resulting initial interface concept design is illustrated in Figure 6.15-4 and leans heavily on the OMV developing configuration utilizing the traversing "SEE". The basic installation and operational differences between the OMV and initial OSCRS interface subsystem is evident in Figure 6.15-5.

TRW's "triangular" facing structure retains the ends of the three-screw translating mechanism and also houses the redundant CCTV's and lights. This facing plate/structure is non-extendable but is removable. The total OMV assembly is an orbital replacement unit (ORU). OSCRS design simply replaces the OMV triangular facing plate with a full circular umbilical plate of essentially the same outside diameter. It includes the same accommodations for the CCTV's and lights as well as the three-screw shaft end supports and translating mechanism. The umbilical plate is fixed to OSCRS supporting structure and is not an ORU.

Further interface definition includes generating and understanding the loads imposed on the OSCRS and spacecraft support structure and the translating "SEE" due to multiple connector engagement/disengagement action. Additionally, transfer separation forces induced by media transfer pressure must be considered. This requires specific connector design/operational data including compliance feature loads. The existing "SEE"/FRGF interface has definitive load limits and are presented in Figure 6.15-6.

6.15.3 STANDARD END EFFECTOR/GRAPPLE FIXTURE

The "SEE" (including the SPAR Aerospace Ltd. modifications) is a hollow, light-gauge aluminum cylinder that contains a remotely controlled motor drive assembly and three wire snares. The drive system provides the ability to capture, rigidize and release a payload. The capture/release function is achieved by a rotating ring at the end of the "SEE" which open and close the wire snares around the

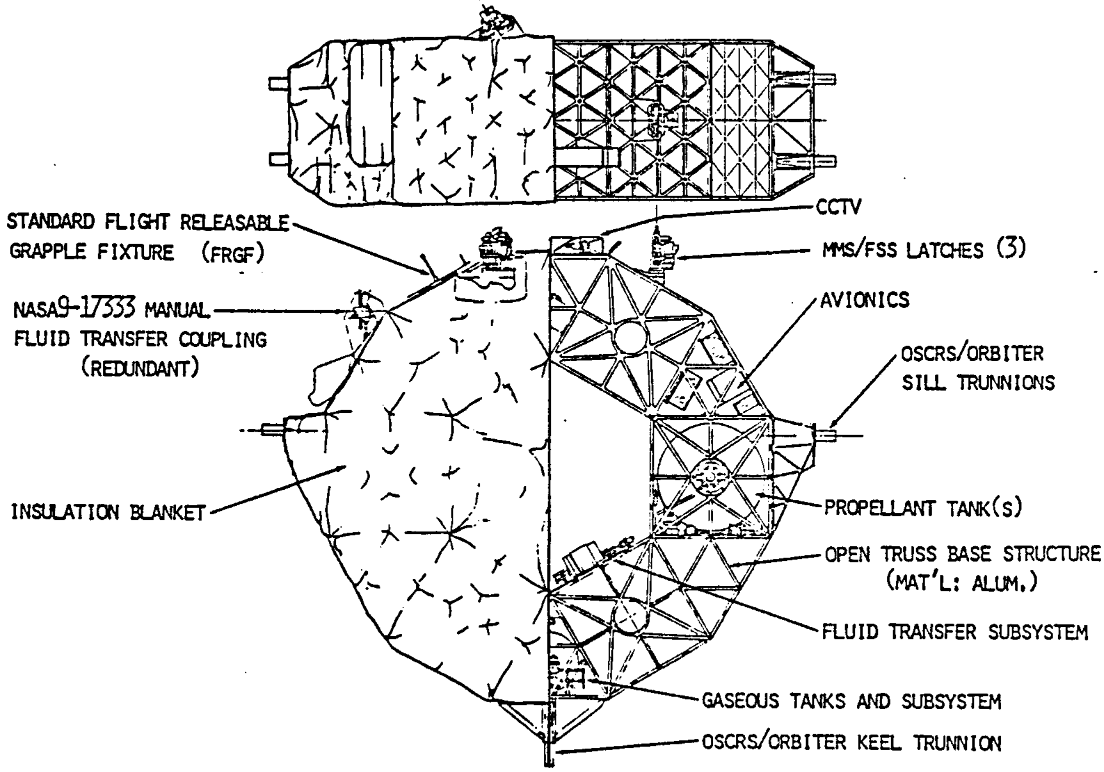


Figure 6.15-1 - Baseline OSCRS Configuration

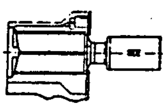
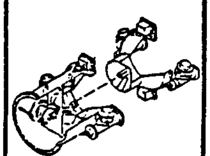
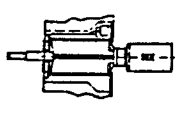
TYPE	ADVANTAGES	DISADVANTAGES
 TRW-OMV / SPAR LINEAR TRAVEL ACTUATOR	O PROVIDES 26.625IN. AXIAL TRAVEL AFTER CAPTURE & RIGIDIZATION BY SEE O LARGE CAPTURE ABILITY (SEE)	O LOW AXIAL LOAD (PULL-IN) CAPABILITY OF 1200 LBF. O CONCEPT ONLY. REQUIRES DESIGN, FAB., DEV.&QUAL. TESTING O REQUIRES SEE/FRGF RIGIDIZING FEATURE FOR 4000 LBF (OSCRS REQUIREMENT)
 TRW-OMV PROBE CONCEPT	O ADEQUATE AXIAL TRAVEL AFTER CAPTURE OF 5.5IN. O HAS RADIAL INDEXING FEATURE	O SMALL CAPTURE AREA (AXIAL ±1.5 IN.) (ANGULATION ±5°) O CONCEPTUAL DESIGN ONLY, REQUIRES DESIGN, FAB.& DEV.,QUAL TESTS
 R.I. MODIFIED SEE (RIGIDIZING LATCHES WITH LINEAR ACTUATOR)	O UTILIZES ST'D. END EFFECTOR O LINEAR/BEARING GUIDE SHAFTS O SIMPLE-REDUNDANT DRIVEN LINEAR ACTUATOR O FEATURES WIDE RANGE OF AXIAL LOAD CAPACITY O MECHANICAL ACTUATED LATCHES RIGIDIZE SEE/FRGF INTERFACE BEFORE UMBILICAL ENGAGEMENT	O CONCEPTUAL DESIGN ONLY, REQUIRES DESIGN, FAB.& DEV.,QUAL TESTS

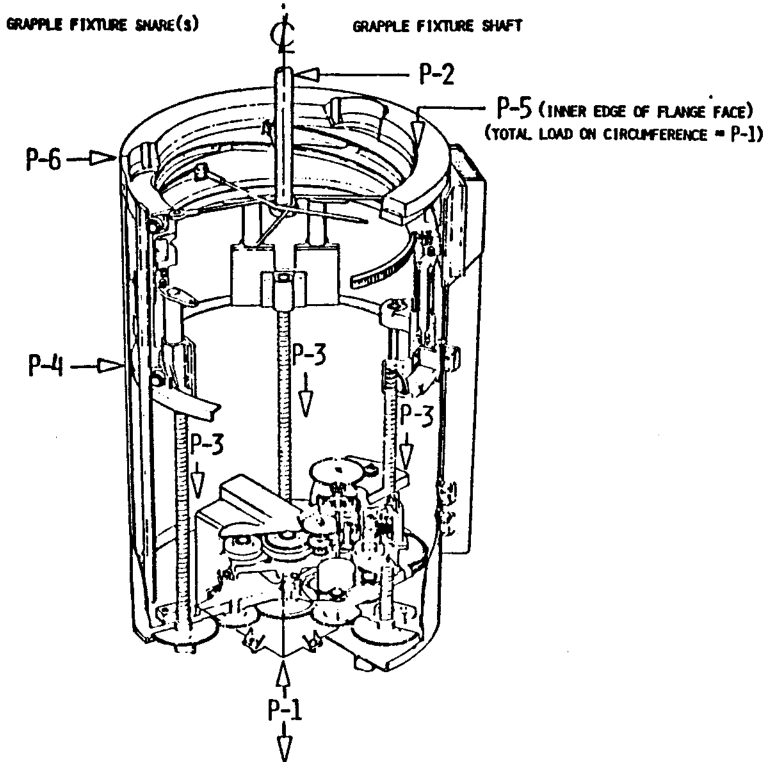
Figure 6.15-2 - Combined Berthing / Umbilical Connect Concepts

payload (spacecraft) mounted grapple fixture. Interface rigidization is achieved when the snare assembly is withdrawn into the end of the "SEE" pulling the spacecraft into full contact with it.

The grapple fixture consists of a long shaft (a rigid shaft for the FRGF and moveable for the rigidize sensing (RSGF) and electrical (EFGF) grapple fixtures), three alignment cam arms, and a target fixture. The rigid shaft, when grappled by the "SEE" snare wires, provide the structural integrity between the OSCRS and spacecraft. Figures 6.15-7 and 6.15-8 illustrate the "SEE" and FRGF, respectively.

The "SEE" operates from a 28VDC supply and is equipped with a passive electrical payload connector (26 pins) as a standard feature used in conjunction with an electrical grapple fixture (EFGF) containing an active mating connector as shown in Figure 6.15-9. A series of brakes and clutches alternately drive and lock the snare and rigidize systems. All loads are applied to the "SEE" and to supporting structure, including the translating mechanism, through the FRGF/"SEE" interface. Torsional loads are reacted through the three cam arms positioned on the grapple fixture. A spring motor provides redundancy to the snare release. The load capability of the "SEE"/FRGF interface is the result of OMV requirements presented in Table 6.15-3.

Very preliminary connector engagement force predictions shown in Table 6.15-2 suggest a marginal condition exists during the consumables transfer operation. A (20) connector umbilical will excert approximately a 2000 lb separation force. The ability of the "SEE"/FRGF to resist separation, due to axial loads, depends on the ability of both the snare ring carriage drive and the translation drive to resist the separation force. Currently, the translation drive can be designed to provide adequate "pull-force" or resistance since the OMV requirements are not yet in "concrete". The weak link lies in the snare carriage drive limitation of 2215 lbf, and is not modifiable by definition. Since the "SEE" is existing qualified hardware, alternative methods must be explored to adequately "rigidize" the "SEE"/FRGF interface.



P-1: 2215 LBS.
P-2: 250 LBS. IN ANY DIRECTION
P-3: 738 LBS. X 3 SCREWS = 2215 LBS.
P-4: 50 LBS. IMPACT LOAD ON OUTER SHELL
P-5: 100 LBS. POINT LOAD (TOTAL LOAD = 2215 LBS.)
P-6: 950 LBS. RADIALLY

Figure 6.15-3 - SRMS End Effector Limit Loads

Table 6.15-1 - Umbilical Connector Count Rational

(100 % REDUNDANT BIPROPELLANT / ULLAGE TRANSFER SYSTEM)

	FUNCTION	NO. REQUIRED	TOTAL
N_2O_4	PRIMARY PROPELLANT TRANSFER	1	4
	REDUNDANT PROPELLANT TRANSFER	1	
	PRIMARY ULLAGE TRANSFER	1	
	REDUNDANT ULLAGE TRANSFER	1	
MMH	PRIMARY PROPELLANT TRANSFER	1	4
	REDUNDANT PROPELLANT TRANSFER	1	
	PRIMARY ULLAGE TRANSFER	1	
	REDUNDANT ULLAGE TRANSFER	1	
HE	PRIMARY TRANSFER	1	2
	REDUNDANT TRANSFER	1	
GN_2	PRIMARY TRANSFER	1	2
	REDUNDANT TRANSFER	1	
AVIONICS	PRIMARY	2	4
	REDUNDANT	2	
ELECT	PRIMARY	2	4
	REDUNDANT	2	
MAXIMUM NUMBER OF CONNECTORS REQUIRED / MISSION			20

(ACCEPTS MONO / BIPROP TRANSFER INCLUDING ULLAGE TRANSFER & 100% DISCONNSCT REDUNDANCY)

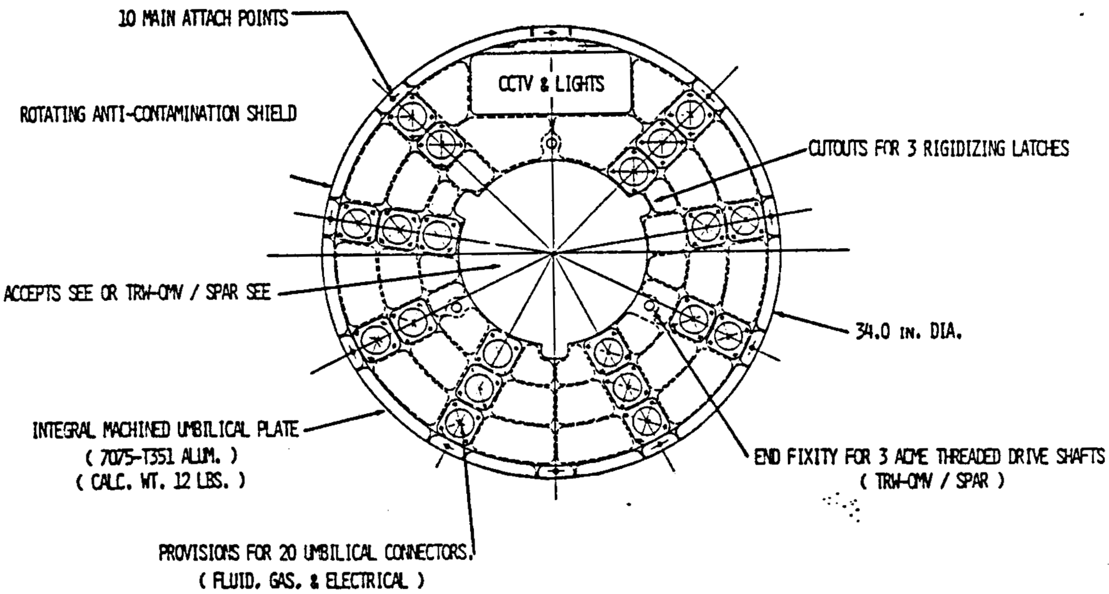
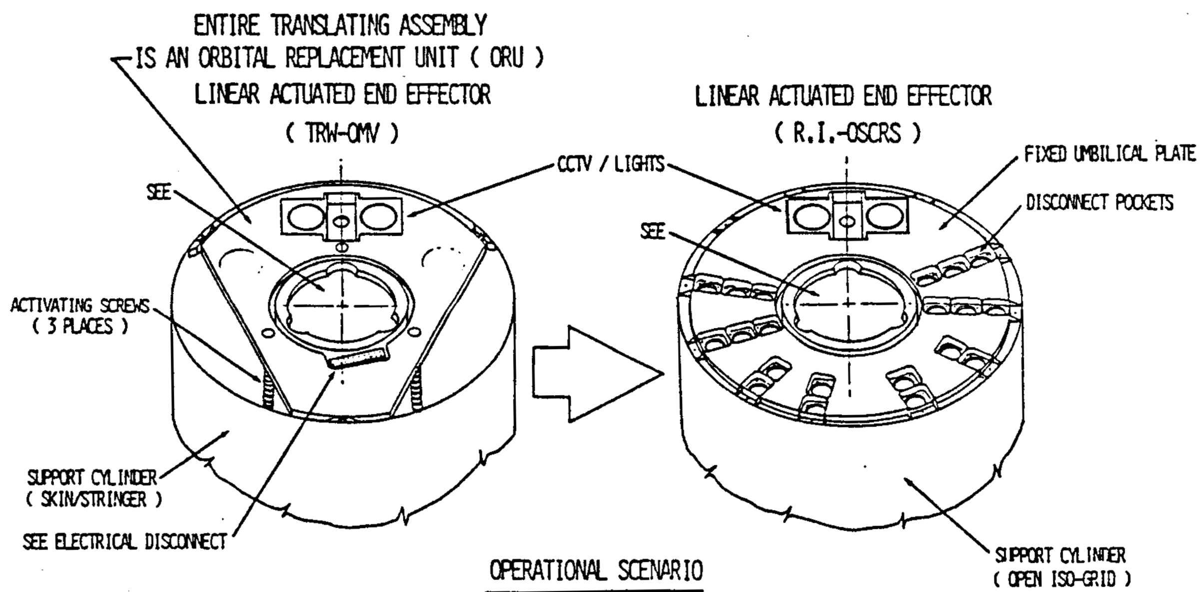


Figure 6.15-4 - Standard Umbilical Plate Configuration



OPERATIONAL SCENARIO

SINGLE ACTUATOR: (1) CAPTURES AND RIGIDIZES S/C
(2) SEE RETRACTS DRAWING S/C INTO CLOSE PROXIMITY OF OSCRS
(3) DURING FINAL ~2 IN OF TRAVEL ALL DISCONNECTS ENGAGE
(4) UMBILICAL DISENGAGEMENT AND RELEASE OF S/C IS IN REVERSE ORDER

Figure 6.15-5 - Common Berthing / Umbilical Engagement

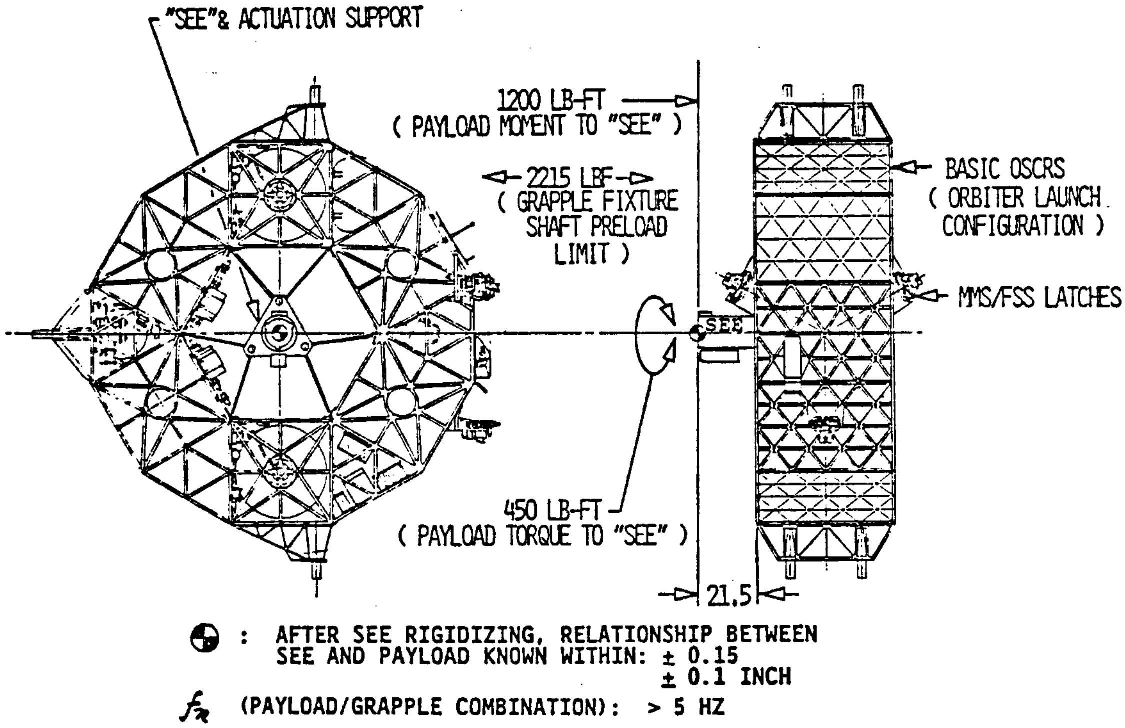


Figure 6.15-6 - Standard End Effector (SEE) Docking Design Requirements

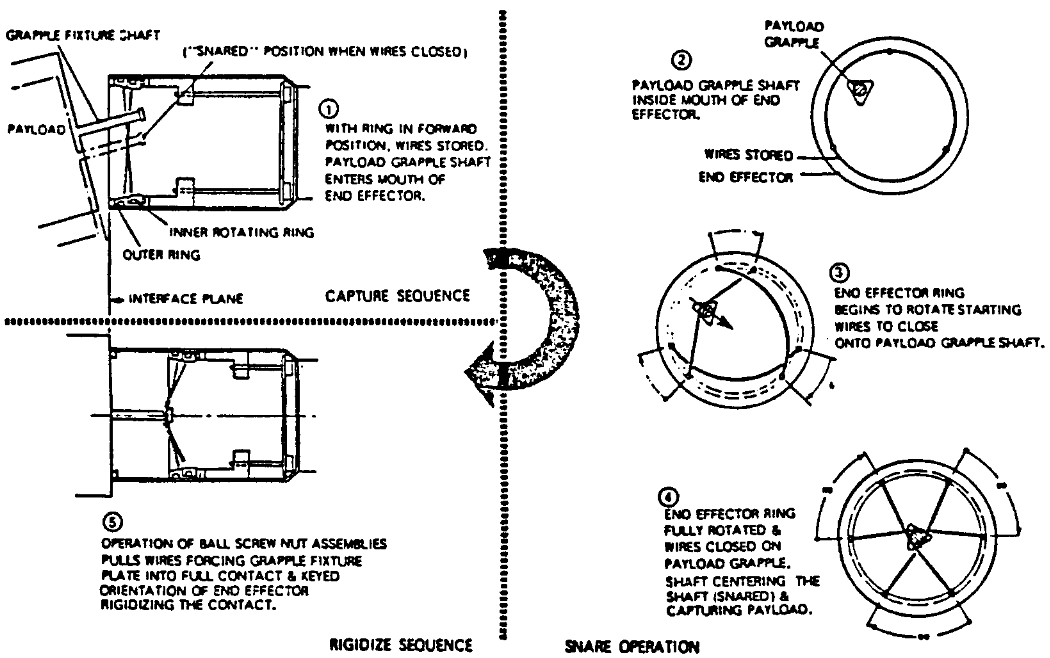


Figure 6.15-7 - Standard End Effector - Capture and Rigidize Sequence

TABLE 6.15-2. UMBILICAL ENGAGEMENT LOADS

(COMMON BERTHING & UMBILICAL ENGAGEMENTS)

- o Bipropellant Transfer Scenario:
  - o N_2O_4 & MMH transfer uses (4) connectors
  - o Ullage transfer uses (4) connectors
  - o 100% connector redundancy for all fluid, gas, & electrical
  - o Electrical uses (8) 40-50 pin connectors
  - o GN_2 & HE transfer @ 4500 MEOP uses (4) connectors

Engagement Forces	Transfer Forces	
(8) Fluid connectors @ 30 lbs ea = 240 lbs	@ 10 lbs ea + 1.1 x =	590 lbs
(4) Gaseous connectors @ 10 lbs ea = 40 lbs	"0 + .13 x 4500 psi =	585 lbs
(8) Elect/Avionics @ 100 lbs ea = 800 lbs	@ 100 lbs ea	800 lbs
1080 lbs		1975 lbs

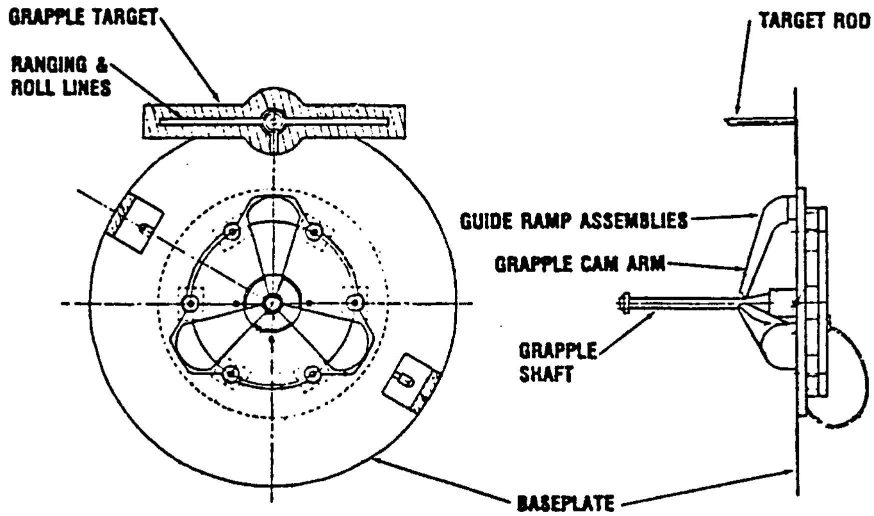


Figure 6.15-8 - Grapple Fixture

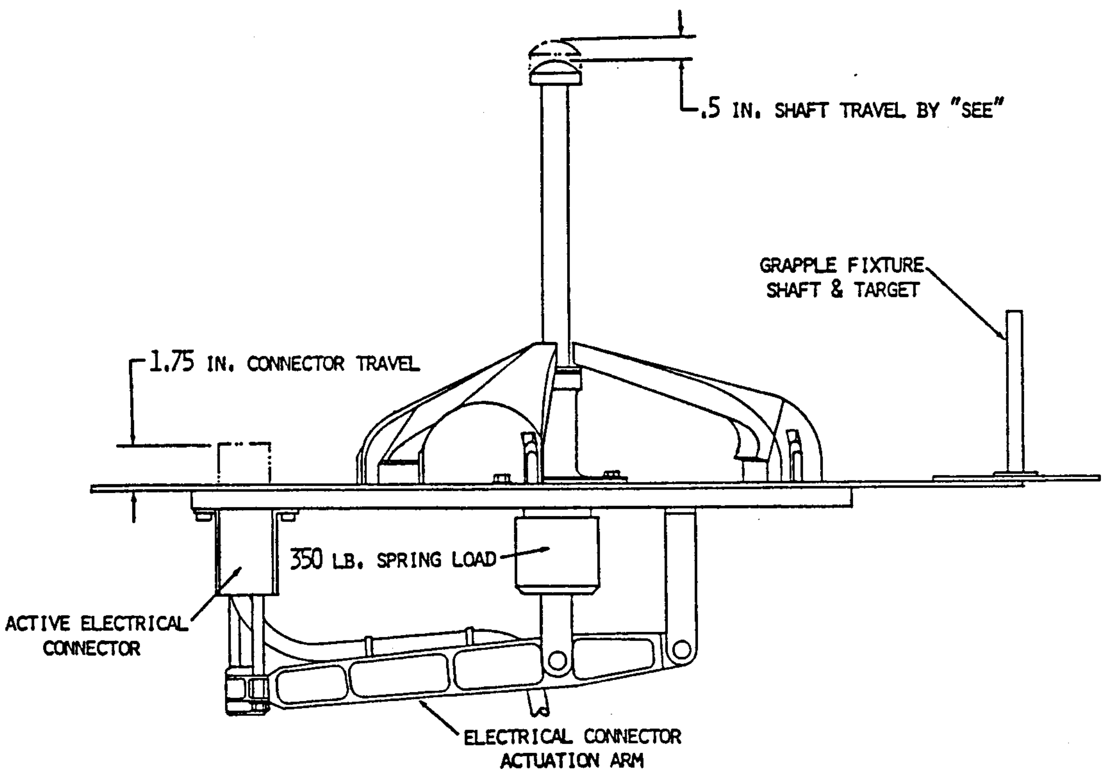


Figure 6.15-9 - Electrical Flight Grapple Fixture (EFGF)

6.15.4 SUPER RIGIDIZATION

Figure 6.15-10 illustrates a rigidizing concept that is simple, light and cost effective. It does not involve additional electro-mechanical devices. Utilizing three pivoting arms located at 120° intervals around the outside circumference of the "SEE", it relies on redundant torsion springs acting about the arm pivot point to maintain the arm in the open position. This allows the FRGF to be drawn into the "rigidized" position by the snare carriage drive. As the translation drive retracts the "SEE", drawing the spacecraft to OSCRS, the pivot rigidizing arm is forced closed over the FRGF base plate creating a vise like grip or "super rigidization" of the "SEE"/FRGF interface.

Sequential adjustment is possible by providing a serrated surface between the supporting clevis and the "SEE" hollow cylinder. Although the clevis interface base could be bonded and/or mechanically attached to the existing "SEE" cylinder, further analysis may indicate that a new "SEE" cylinder with integral clevis bases at three places may be a more practical approach. The super-rigid interface overcomes the axial limitations of the "SEE"/FRGF interface during umbilical engagement and consumables transfer forces. Only the translation mechanism (TRW-OMU) pull-forces need be examined for possible modification.

6.15-5 SUPPORT STRUCTURE

The common berthing/umbilical interface concept involves: (1) both strength and rigidity against engagement/disengagement forces and, (2) accuracy in location for mating connector halves. Both demand rigid non-flexing structure. The support structure physically integrates all the other berthing/umbilical subsystems.

All the structural elements must be made as large as possible. This minimizes local fittings and reinforcements which add little to stiffness but much in cost and weight. Anticipating access requirements, an open structure (truss) design is needed. Current aerospace structural design emphasizes the triangulation approach since triangular arrays of bars can carry all types of loads without skin and can stiffen skins to function more efficiently. Open truss structure can be easily machined from single plates on numerical control mills at a fraction of the cost of built-up skin/stringer assemblies. Open truss structure can be produced with more equivalent strength compared to sheet metal and be 13 percent lighter for equivalent area.

Applying this basic philosophy, Figure 6.15-11 depicts the completed common berthing/umbilical assembly installed in the baseline OSCRS structure with very minimal modification to the original structure. Figure 6.15-12 is an exploded view revealing the major parts.

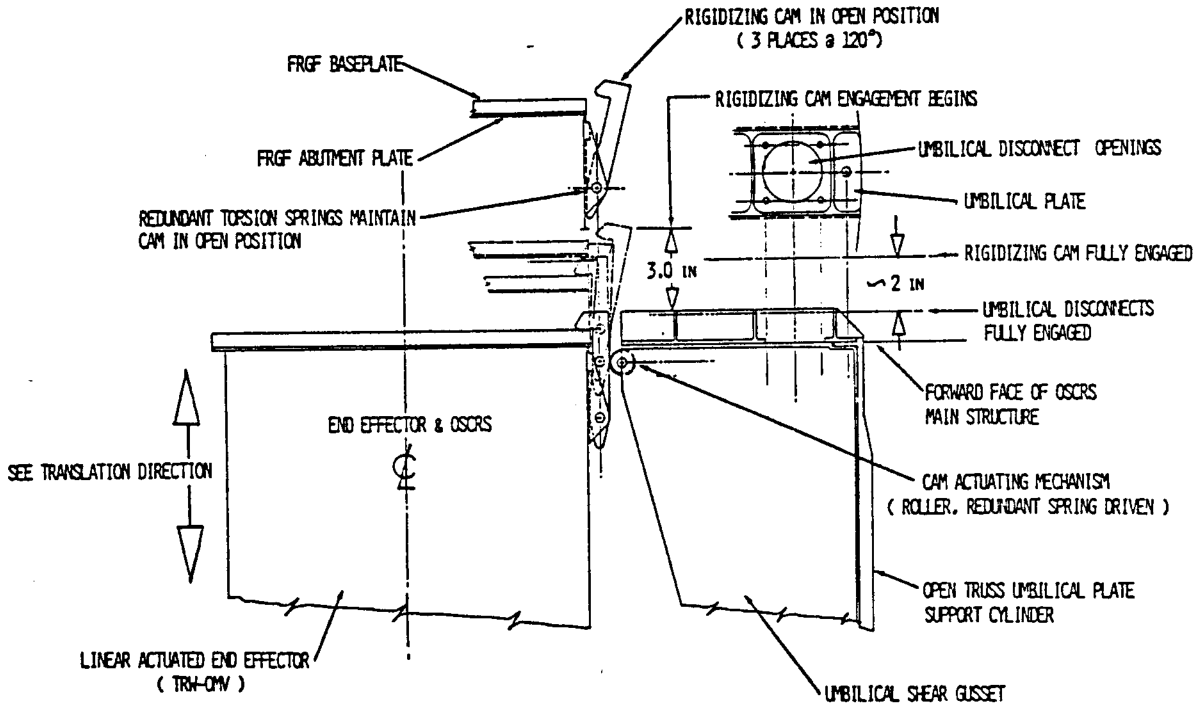


Figure 6.15-10 - SEE / EFGF Rigidizing Concept

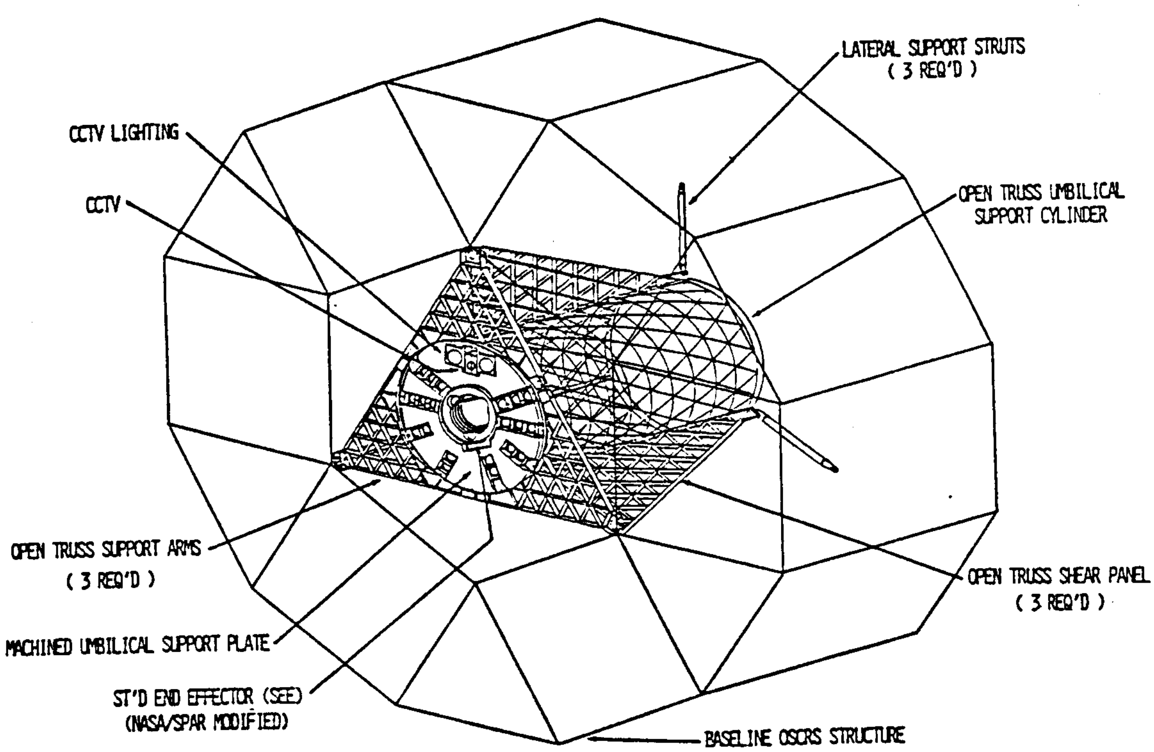


Figure 6.15-11 - Combined Berthing / Umbilical Configuration

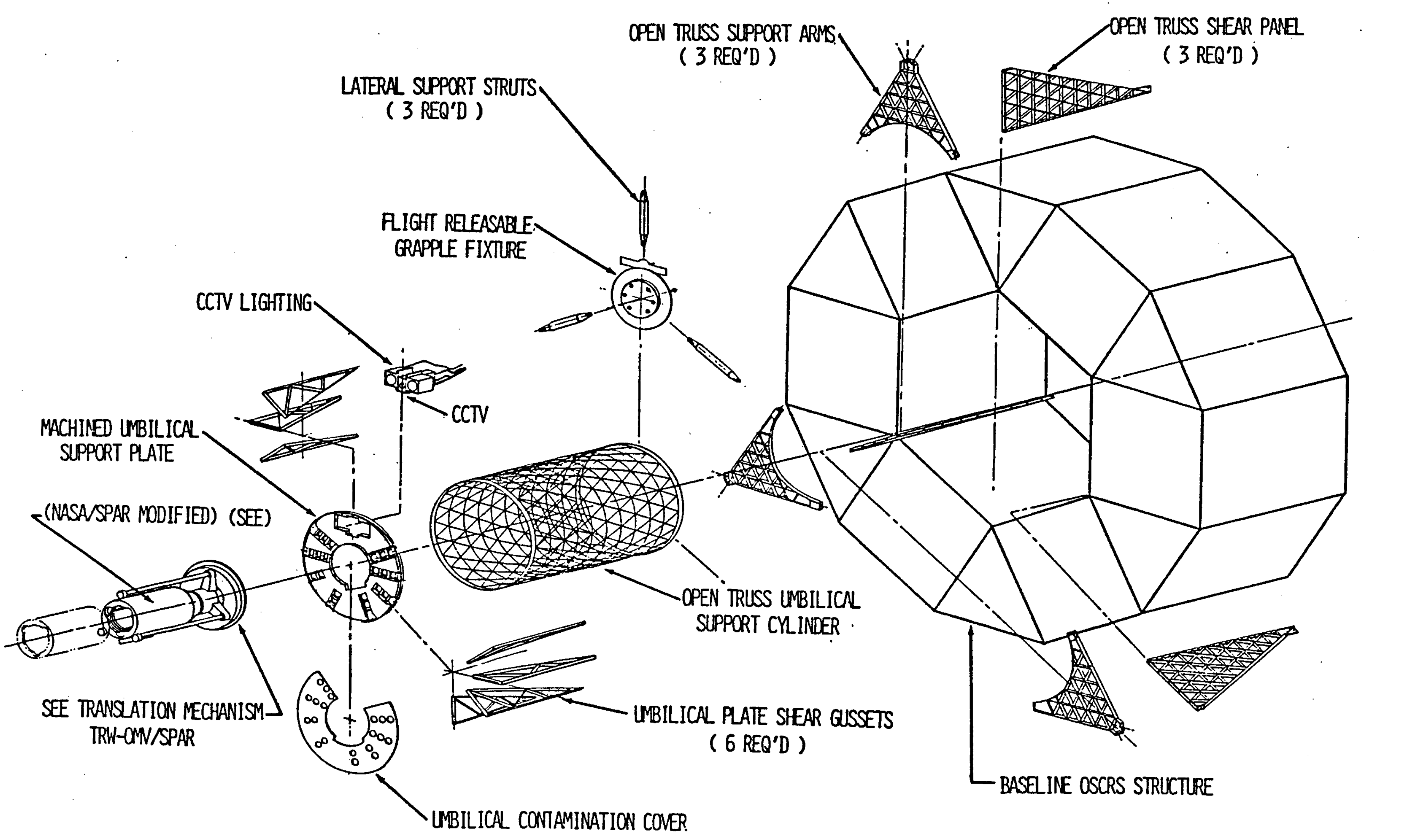


Figure 6.15-12 - Combined Berthing / Umbilical Configuration

Although the integrally machined umbilical plate will possess inherent stiffness by configuration, resistance to "oil-canning" is provided by six identical umbilical plate open truss shear gussets. Mounted at one end of the open truss support cylinder is the umbilical plate assembly and related equipment. At the opposite end a grapple fixture with lateral support struts (aluminum tubes) completes the structural assembly.

From a mission peculiar standpoint, both the grapple fixture and the umbilical plate can be easily replaced with alternate configurations.

Table 6.15-3 presents a calculated structural weight statement based on the described configuration.

6.15.6 ALTERNATE CONCEPTS

Alternate concepts have been investigated. Figure 6.15-13 illustrates a concept that differs from the point design of this study inasmuch as it features sequential berthing and umbilical engagement utilizing MMS/FSS latches and a translating umbilical plate.

This concept is shown installed in two locations on the baseline OSCRS structure. One is located centrally in the hexagonal open area. The other is positioned on the outer 12-sided polyhedron perimeter. Both concepts rely on a translating umbilical plate (circular on rectangular) motivated by a linear actuator and guided by Sarrut links which permit only straight line translation.

Both concepts suffer from the same limitation in that they must rely on a separate device other than a translating "SEE" to provide berthing. This increases both weight and complexity. The outer perimeter location would, however, be an acceptable approach when the OSCRS is located in the orbiter payload bay. By utilizing the orbiter RMS, spacecraft can be berthed on OSCRS as envisioned originally for the gamma ray observatory (GRO). The centrally located position might be adaptable to mating with the space station using a similar RMS for capture/positioning.

Of the alternate concepts analyzed, none exhibit the flexibility or operating simplicity of the proposed common berthing/umbilical concept presented in this study.

6.15.7 UMBILICAL DISCONNECTS

Finalization of the modified "SEE" translating mechanism axial force requires definition of the actual load encountered across the umbilical. This requires detailed disconnect data relating to engagement/disengagement forces as well as compliance and transfer pressure separation loads.

ITEM	QUANTITY	TOTAL IN^2	UNIT WEIGHT ① (LB/IN^2)	WEIGHT/LB
CYLINDER	1	1409	.015	21.1
SUPPORT ARMS	3	884	.024	21.0
SHEAR PANELS	3	1716	.0285	48.9
SHEAR GUSSETS	6	1560	.0285	44.5
UMBILICAL PLATE	1	935	.013	12
				147.5
ATTACHING HARDWARE	AR		10%	14.8
STRUCTURAL CONTINGENCY			5%	7.4
				TOTAL 169.7 LBS

① (MEAN $\bar{t}$ = 0.282" ;~23% TOTAL ;~ 4.10 PSF)

Table 6.15-3 - Structural Weight Statement

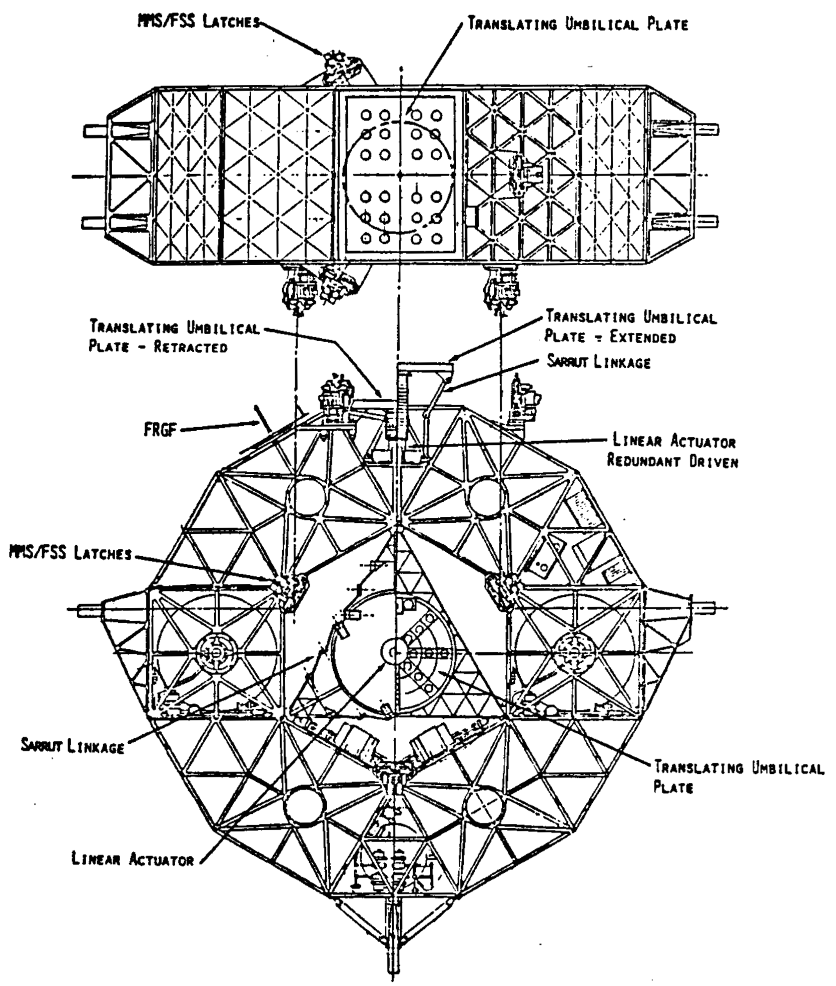


Figure 6.15-13 - Separate Berthing / Umbilical Concept

LOW PRESSURE (LIQUID/GAS)

Surveys reviewing off-the-shelf space qualified umbilical connectors failed to identify a candidate valve meeting all the desired characteristics listed in the selection criteria found in Table 6.15-4. A sample of the disconnects reviewed included:

- o Snap-Tite 29
- o Fairchild Controls Co. - LEM
- o Seaton-Wilson 2-660
- o Wiggins 6000 Series
- o NASA P/N 76300000-101

TABLE 6.15-4. UMBILICAL DISCONNECT REQUIREMENTS

(3)	Envelope Diameter	2.0 inch (max)
	Weight (each coupling half)	2 lbs (max)
	Operating Pressure	500 psia (MEOP)
	Proof Pressure	1000 ± 10 psia
	Burst Pressure (without burst)	2000 psia (min)
	Internal Compliance Features	Lateral offset: ± 0.1 inch. Angulation: ± 5° (min) Axial Overtravel: ± 0.25 inch
	Pressure Drop	15 psia of H_2O @ 500 GPM
	Redundant Seals - Two Failure Tolerant	
	Engagement/Disengagement Forces	30 lbs (max)
	Spillage Volume	1.0 cm^3
	External Leakage (Disconnected)	1 x 10^{-6} SCCS (GHE @ 500 5 psi)
	External Leakage (Connected)	
(1)	Media Compatibility	GHE, GN_2, H_2O, N_2H_4, MMH, N_2O_4
	Operating Temperature	0°F to 250°F
	Transfer Pressure Separation Forces	1.1 (factor) x MEOP/ Connector
(2)	Line Connection Configuration	St'd tube stub (weld/braze)
	Repairable Weld Construction	

(1) Different transfer media require different materials
(2) Connector configuration shall prohibit cross installations/usage
(3) Mounting provisions to be outside of this diameter and are (TBD)

Of the criteria listed in Table 6.15-4, the internal compliance requirement was ranked second only to reliability. Internal compliance significantly reduces the total system weight and cost. A balanced pressure design and minimal pressure drop as well as low engagement/disengagement forces were also considered very important. The ability to disengage without involvement of collets, collars, latches, balls, etc., was considered mandatory.

Early during the OSCRS study, Moog Inc. Space Products Division, East Aurora, New York, proposed a new connector concept to industry. The design, a small umbilical subsystem, designated Model 50E 559 AUC, consisted of two model 50E 565 RSO connectors combined into a system demonstration unit. This unit featured fully automatic, remote operation. It utilized an electro-mechanical linear actuator engaging/disengaging two valve halves. Each valve half featured a rotary ball shut-off sequentially operated mechanically during the engagement stroke. Minimum spillage at disengagement, minimal ΔP, multiple seals and compatibility with the common gases, water, and hydrazine (N_2H_4) were the salient features. The valve did not feature internal compliance.

Figure 6.3-4 illustrates the Model 50E 565 SRO disconnect and the sequential ball rotation before, during and after engagement. Moog Inc. demonstrated the complete system operation by video tape. The system worked well.

Based on the Rockwell opinion that a successful connector must provide internal compliance, Moog Inc. proposed dwg. A99177, (shown in Figure 6.15-14) This concept featured all internal compliance, except axial, which will be included in a new model concept.

The umbilical plate configuration defined in this study is based on a connector design complying with the requirements in Table 6.15-4.

<u>HIGH-PRESSURE (GAS)</u>

High pressure (1500-4500 PSIA) helium and nitrogen connector requirements differ significantly from low pressure connectors ($<$ 500 PSIA).

Internal compliance bellows, particularly with L/D ratios greater than 1.0, are unstable at high internal pressures unless laterally constrained. Unconstrained, they react similar to a slinky toy, that is, they squirm. Bellows should not be used in this application.

As in low pressure connectors, no latching of connector halves after engagement would be accepted.

Small area differentials, across connector interfaces are essential to minimize internal separation forces since the transfer pressures are high.

ELECTRICAL/AVIONICS

Multiple pin electrical connectors, both manual and remote, are available as qualified, off-the-shelf components. Most remote connectors however, are manually connected (STS orbiter-external tank accepted) and disconnect when the opposing sides of the interface separate. There is no such thing as a standard electrical connector. Even modest changes in shell configurations consistent with mission peculiar mounting requirements will result in new hardware/identifications.

A typical connector used between the external tank and Orbiter is shown in Figure 6.15-15. This 55 pin connector simply slides apart as the two vehicles separate. Special shell designs incorporating mounting provisions applicable to the standard umbilical interface recommended in this study (including compliance features) will be required.

Pin engagement overtravel must be included in all connectors as the internal axial compliance feature. Hermetically sealed interfaces should not be required thus simplifying overtravel design.

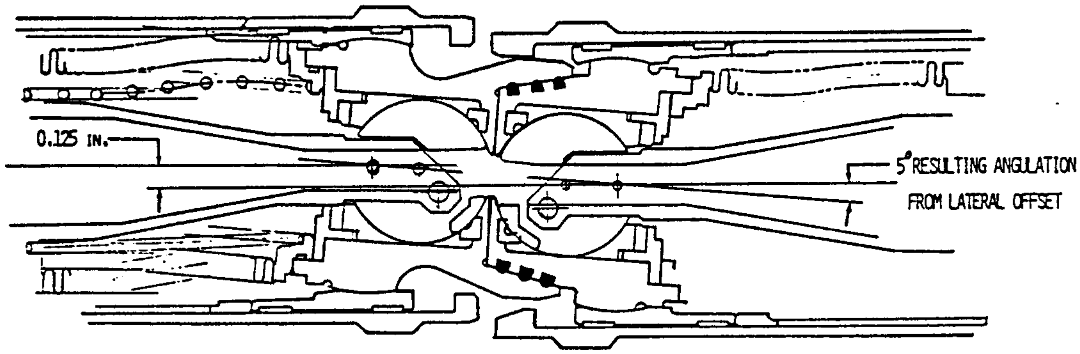
6.15.8 A POINT DESIGN (UMBILICAL PLATE)

A point design shown in Figure 6.15-16 includes refinements to the umbilical plate shown in Figure 6.15-4. The umbilical positions have been revised to include 20 positions in two concentric rings surrounding the "SEE" assembly. This provides room to accommodate the three rigidizing links shown in Figure 6.15-17.

The umbilical plate is machined from a 1.5 inch thick 7075-T351 aluminum plate. Cutouts provide clearance for SPAR Aerospace Ltd's new position switch box assembly as well as the "SEE" electrical connector. The "SEE" connector is passive (ie, the electrical engagement translation is provided by the EFGF) and in most missions probably would not be used. Positions are available on the machined umbilical plate for electrical connectors. The EFGF is larger and heavier than the FRGF and is carried by the spacecraft; thus, the electrical connector location on the umbilical plate is preferred.

RIGIDIZING ARMS

Operational details of the rigidizing arms remain unchanged from those discussed earlier. An emergency release mechanism has been added for redundancy release. Figure 6.15-17 shows this release addition. It provides grapple fixture/spacecraft separation in the event of a failure of the "SEE" to translate the spacecraft away from the OSCRS thus releasing the rigidizing arms. The emergency release design



DISCONNECT FULLY ENGAGED WITH 0.125 IN. LATERAL OFFSET

Figure 6.15-14 - Moog Fluid / Gas Disconnect

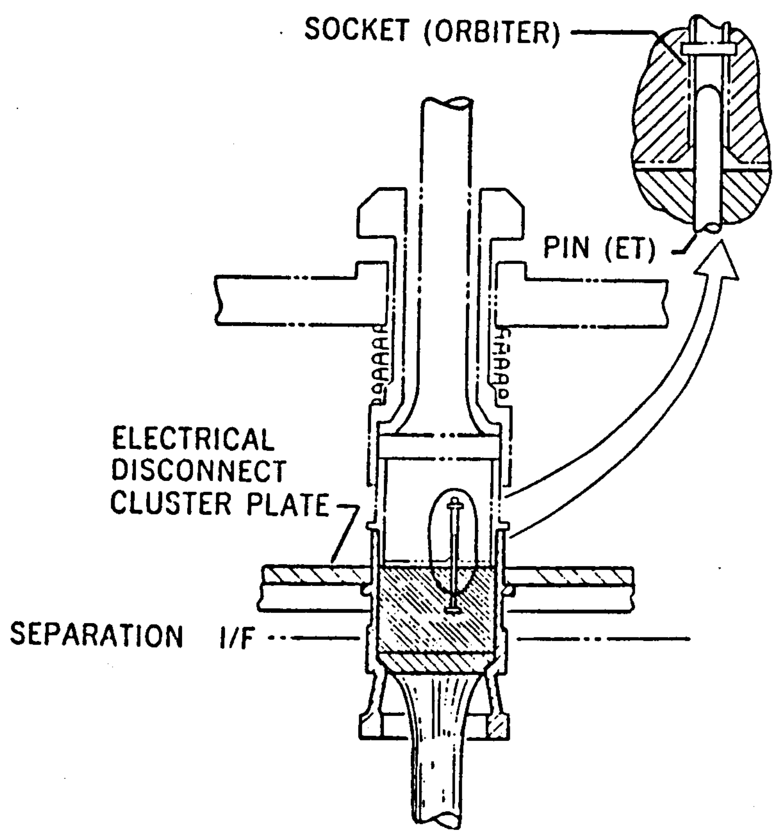


Figure 6.15-15 - ET / ORB Electrical Disconnects

C-2

utilizes redundant NASA standard initiators (NSI) to generate gas pressure moving the emergency release piston away from the rigidizing arm release sear. This permits the redundant torsion springs to open the rigidizing arm permitting release of the grapple fixture/spacecraft from the "SEE" through operation of the snare carriage drive unit. A redundant spring motor inside the "SEE" is available in the event of snare carriage drive failure.

6.15.9 CONTAMINATION CONTROL

A significant feature of the proposed common berthing/umbilical concept involves contamination control. Maximum external leakage (connector separation spillage volume) is defined in Table 6.15-4 as less than 1.0 cubic centimeter. This represents a liquid sphere of 0.488 inches in diameter. Assuming this sphere is MMH or N_2O_4 and is spilled on a wetting surface under 1G and less than 70^oF and puddles 0.06 inches deep, it would form an area approximately one square inch. This is not a large area. In space, conditions are quite different. The pressure is essentially zero and temperatures low. Under such conditions the liquid spillage will almost instantaneously flash-off into a form of snow particles and most probably will wet the immediate surfaces, including the connector from where it came. In a short time period it should evaporate.

Without lateral motivation forces, wide dispersion seems remote. At any rate, the space environment (low pressure/temperature) would not support reaction between the hypergolic propellants.

One solution is to separate the umbilical connectors as widely as possible. Spacecraft, particularly smaller satellites, present limited areas in which to mount connectors. This problem is amplified if connector redundancy is required. Common sense, however, dictates as wide a separation as possible.

An additional form of contamination protection is to provide covers for the umbilical connectors. The covers in the point design actuate by sliding on the interface side of the umbilical plate. The cover is semi-circular ("c"-shaped) in shape with oversized clearance holes appropriately spaced to permit passage of the spacecraft connector for engagement to the OSCRS connector mounted under the umbilical plate.

The covers do not offer protection from connector interface separation spillage. It would be effective in confining small seal leakage and aid in prohibiting foreign substances from adhering to the seals and sealing faces prior to connector engagement.

In conjunction with cover plate protection, axially-staggard installation of spacecraft connectors for different fluids adds significantly to safe disengagement operations by guaranteeing that one media connector is covered before the opposing connector disengages.

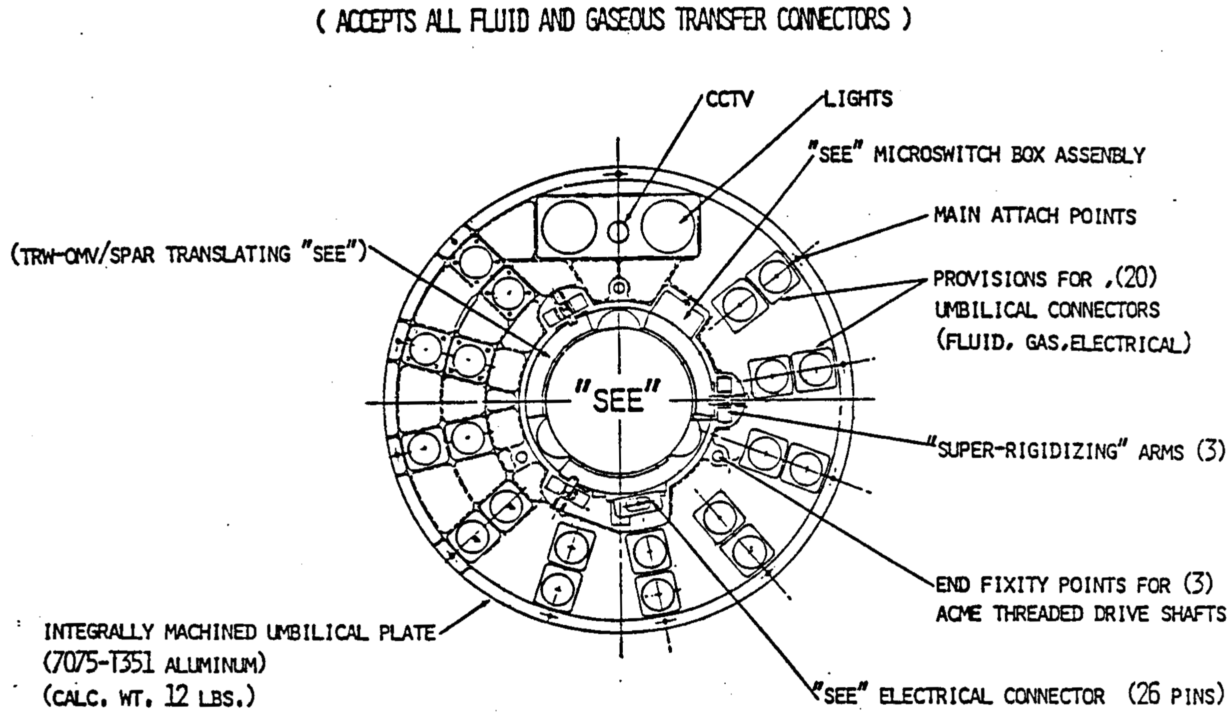


Figure 6.15-16 - NASA Standard Berthing / Umbilical Plate Configuration

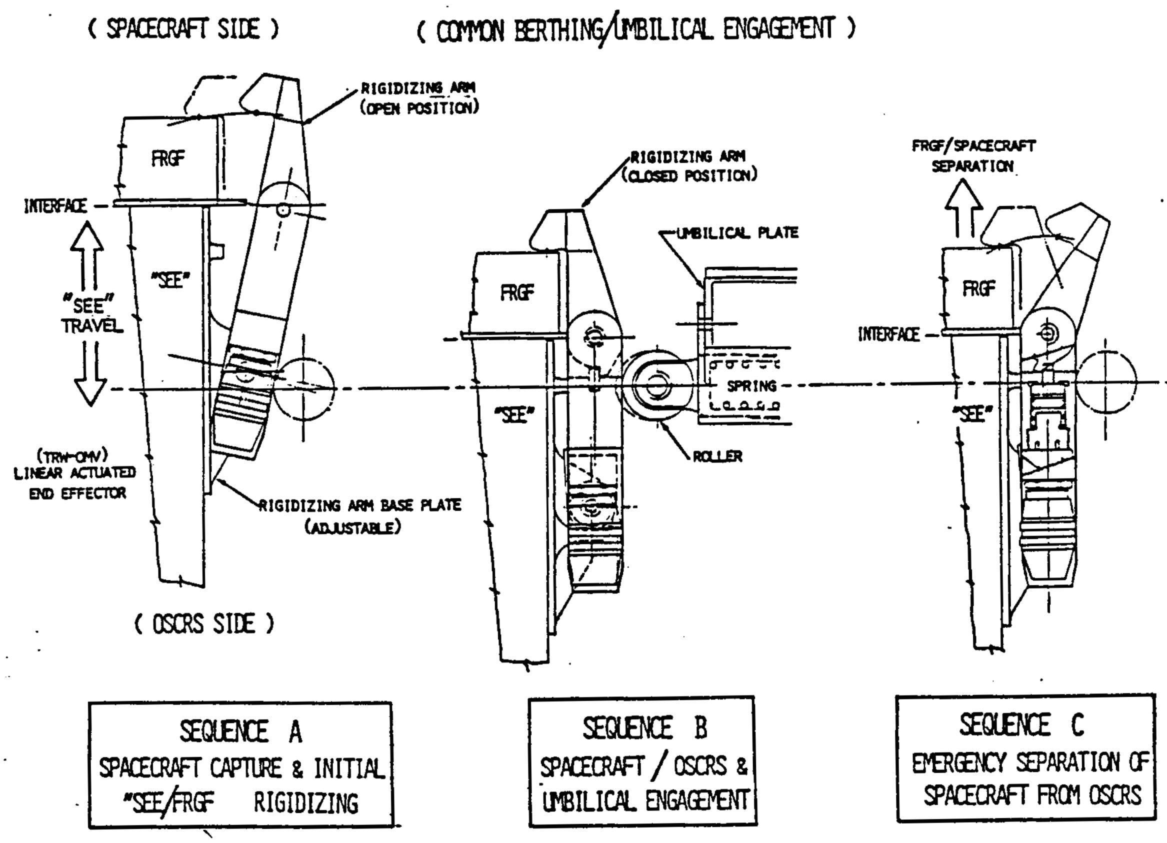


Figure 6.15-17 - SEE / FRGF Super- Rigidizing Concept

From these analyses the final umbilical requirements for contamination control and safety can be stated:

1 - Maximum separation between connectors transfering different fluids, consistent with the limited space available on many spacecraft.

2 - Staggard spacecraft connectors transfering different fluids, resulting in sequential connector engagement/disengagement.

3 - Contamination covers for all connectors.

After spacecraft capture and initial "SEE" rigidizing, the spacecraft is pulled toward the OSCRS umbilical interface. At a predetermined distance from the interface "super-rigidization" occurs as the three rigidizing links pivot over the spacecraft grapple fixture locking it to the "SEE". Small axial allowance must be made between the grapple fixture and "SEE" faying interface to assure the rigidizing links will operate. This slight, ~ .010/.020-inch of axial "play" should not effect the rigidity of the system for accurate connector engagement.

Figure 6.15-18 sequentially illustrates the engage/disengage action. The upper connectors, 1B and 7A are fixed/mounted to the spacecraft in a staggard relation (axially) to each other. The lower connectors, 1D and 7C are mounted, in plane to the OSCRS umbilical plate. Connector 1D is fixed and 7C is allowed to traverse axially through its internal compliance.

Figure 6.15-19 illustrates the preferred method for operating the contamination covers.

A single two stage rotating cam is fixed to a shaft containing a small pinion gear. A rod, containing a roller on the one end, located tangentially and normal to the cam/gear shaft and possessing an integral gear rack, drives the pinion gear and rotates the cover cam. The translating rod/rack is actuated by an adjustable cam mounted on the side of the translating "SEE". This cam controls the timing of the cover cam to the travel of the disconnects. The covers return to the closed position using redundant spring force. The contamination covers are contained in plane by an integrally machined rib in the umbilical plate. Containment of the covers to the plate surface is done by washers fastened to the umbilical plate rib at strategic intervals.

6.15.10 THERMAL

Thermal requirements pertaining to the umbilical connectors are, by nature, best resolved by the respective interface owners. An examination of thermal considerations resulted in a conceptual design shown in Figure 6.15-20.

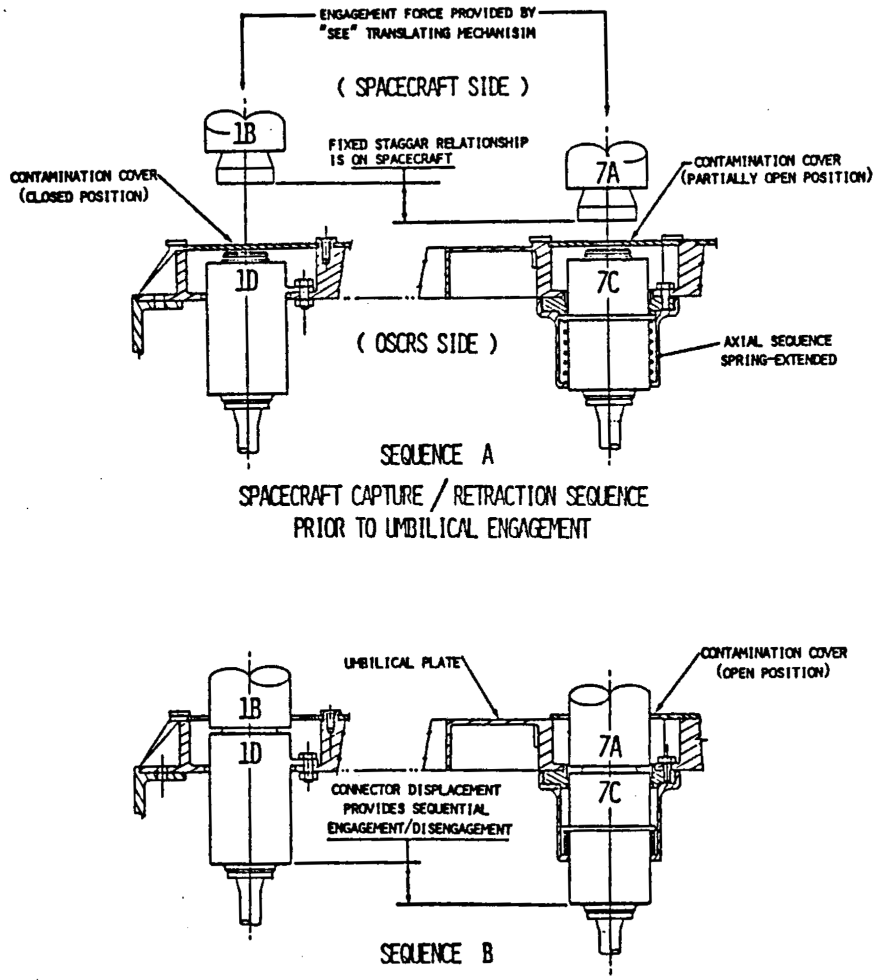


Figure 6.15-18 - Umbilical Engagement Sequence

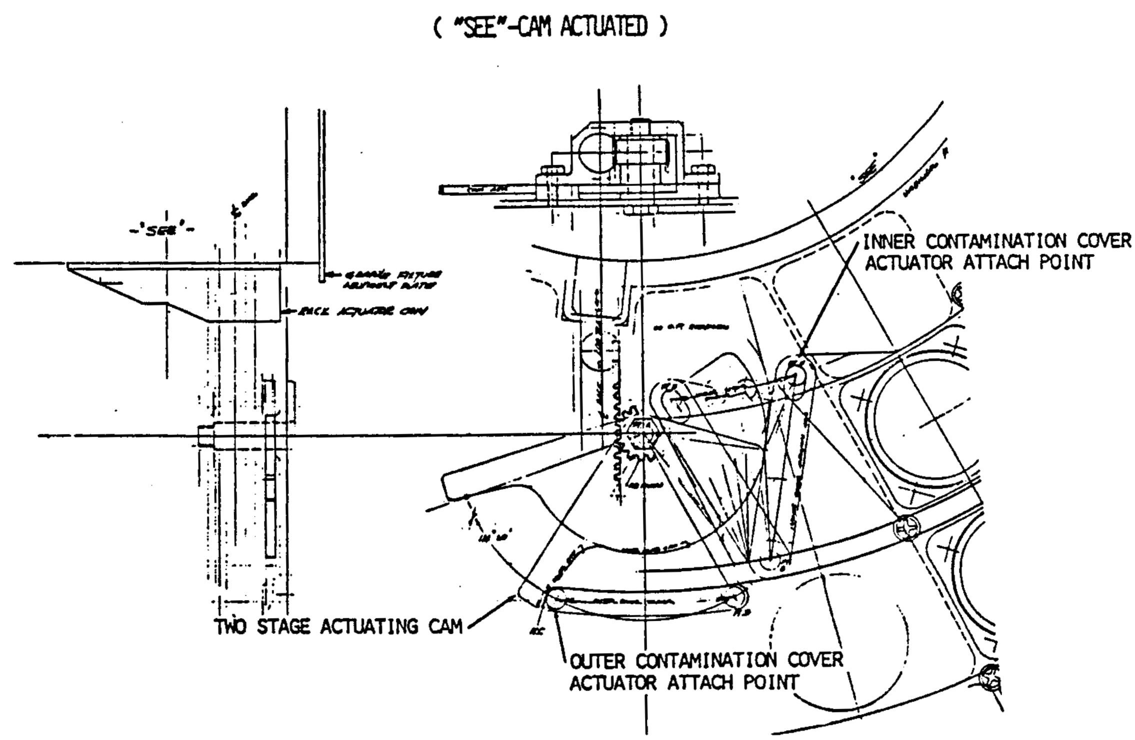


Figure 6.15-19 - Umbilical Connector Contamination Cover Actuation Concept

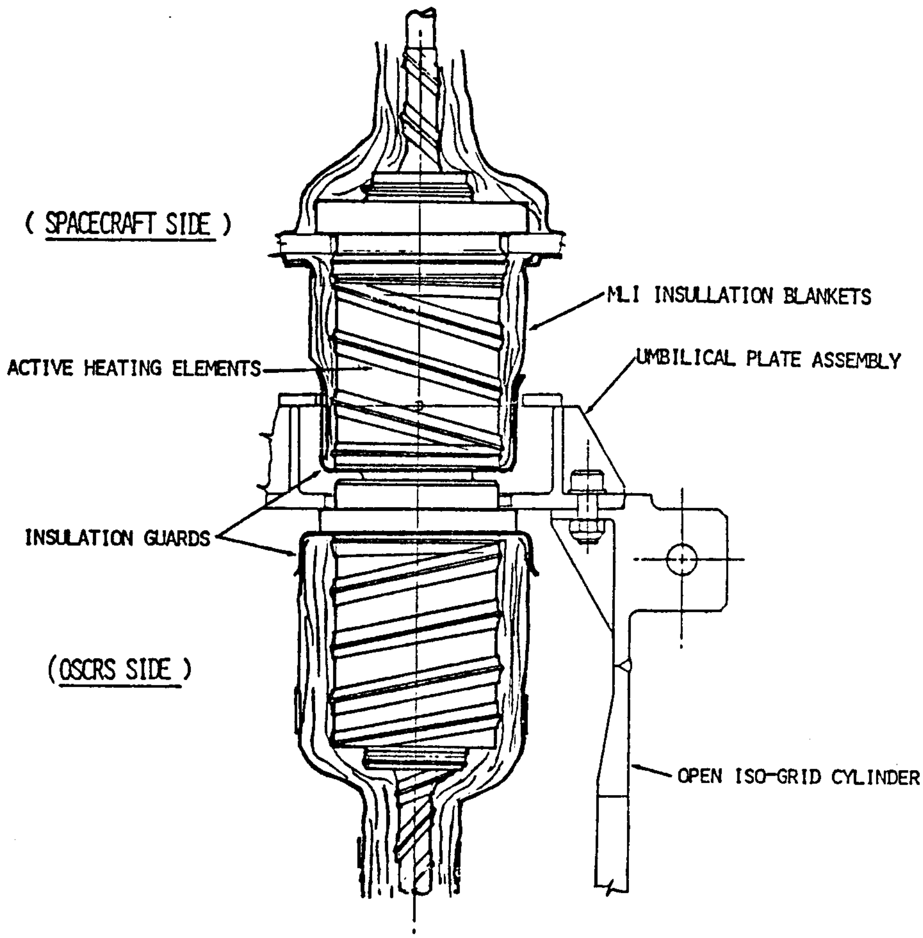


Figure 6.15-20 - Thermal Environment Control

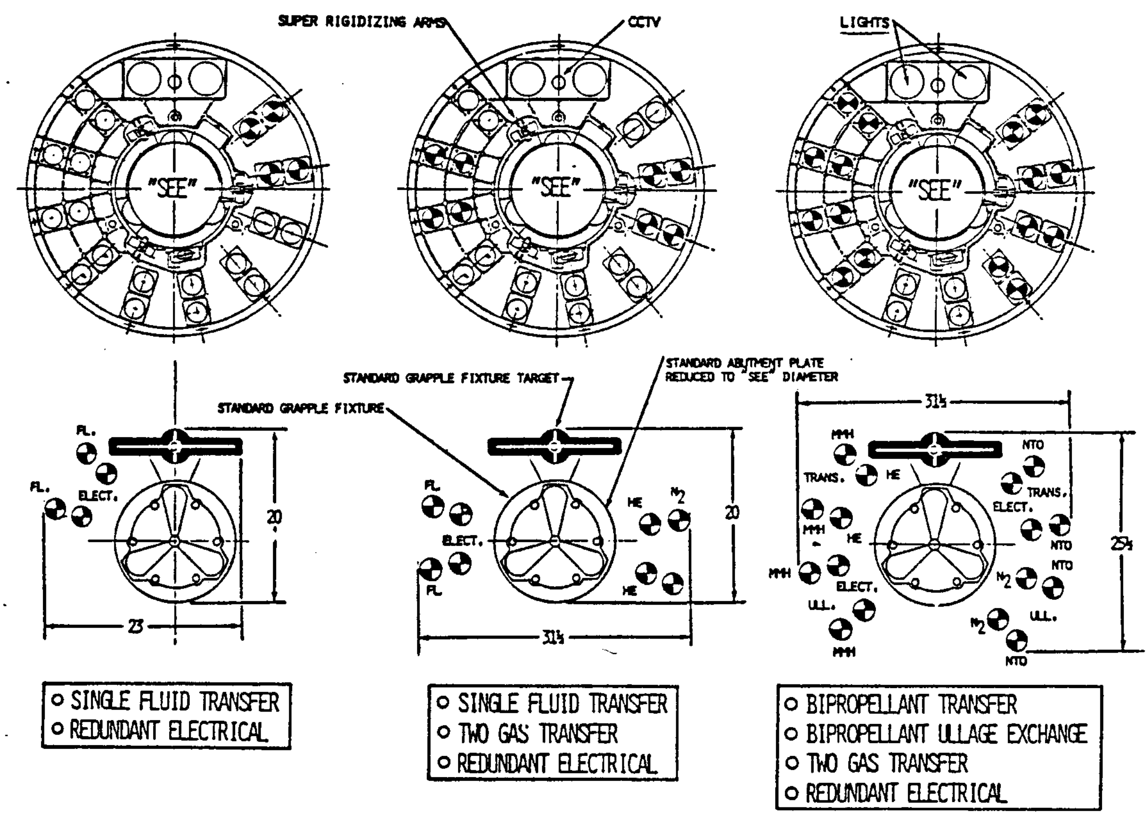


Figure 6.15-21 - Typical OSCRS / Spacecraft Interfaces

The present and proposed modified "SEE" is actively thermally protected by heater elements applied to the outer aluminum cylinder. This is covered by insulation and an outer "Beta" blanket. The thermal control protects the internal actuating electric motors. The addition of the three external super rigidizing arms and supports to the "SEE" cylinder would require additional local cutouts in the outer blanket. The added arm assemblies would be painted white.

6.15.11 CONCLUSIONS/RECOMMENDATIONS

The level of hazards involved in disengaging hypergolic propellants, even remotely and sequentially, is potentially greater than that for mono-propellants. Wide separation between dissimilar fluid connectors, contamination covers, and sequential disengagement as well as transfer line venting and possible purging should reduce the hazards to an acceptable level.

To maximize the universality of a standardized interface a concept featuring a common berthing/umbilical interface that included spacecraft capture, rigidizing, retrieval and umbilical engagement using an integrated, common actuating mechanism has been proposed using existing qualified hardware, where applicable. Figure 6.15-21 presents three possible mission peculiar umbilical interface connector arrangements based on this study's concept.

The major hardware for the point design is either available (i.e., NASA's "SEE") or in development such as TRW's OMV translation mechanism and Moog's umbilical connectors. Design and development work is required for adaption of space-qualified electrical connectors to the common berthing/umbilical concept. Additionally, micrometeroid protection must be addressed.

A clearer understanding of the results of disconnecting hypergolic connectors in the space environment is required. A simple demonstration test in a 707 "Zero-G" simulated flight would be an economical approach. Assembly disconnection of several disconnects (with varying spacing and using actual propellant) in an evacuated chamber, could be accomplished during the 35 seconds of weightlessness with camera coverage.

It is recommended that the NASA "spin-off" the berthing/umbilical interface requirements from further involvement with resupply vehicle studies. A development program should be started initially defining an interface concept. The user community's input and concurrence should preceed final design, test and evaluation. This is a minimal, low cost program with high potential return to the entire related space industry and the need is now.

6.16 OSCRS Launch via an ELV

Launching an OSCRS into LEO in support of supply/resupply missions will become a reality in the near future. STS orbiter cargo manifesting is and will continue to be at a premium. Relief must be forthcoming relative to OSCRS since frequent OSCRS launches will be a necessity. Launching an OSCRS into a LEO parking orbit on an ELV, rather than in the STS orbiter bay, was examined as a feasible concept.

The two primary parameters involved in determining the ELV launch requirements are payload weight and envelope. Figure 6.16-1 presents the OSCRS overall envelope. The baseline resupply schedule, as discussed in section 6.4 assumed four resupply flights per year. Flight 1 carried three tanks of hydrazine totaling 3,720 lbs. Flights 2 and 4 resupplied 3,870 lbs of consumables each and the third flight 5,760 lbs. A total consumables transfer requirement carried by OSCRS for the four flights total 17,280 lbs. The actual single mission capacity of a six GRO tank OSCRS bipropellant resupply configuration totals 8,710 lbs., including 20 lbs of helium and 140 lbs of nitrogen. For gaseous consumables, 36 cylindrical bottles, arranged in a cascade system, are required. Adding the balance of support systems, a single OSCRS weight would total 11,900 lbs.

Table 6.16-1 lists existing and in-development domestic ELV's and their respective payload capacity for a LEO launch. From Table 6.16-1 it is obvious that only two TITAN vehicles offer sufficient launch capability along with growth potential. Figure 6.16-2 presents the selected ELV configuration/capability.

TABLE 6.16-1. ELV PAYLOAD CAPABILITIES

EXISTING U. S. ELV'S	PAYLOAD TO LEO (LBS
DELTA II	7,610
ALTLAS CENTAUR	12,300
TITAN 34D	32,800
TITAN IV	38,600

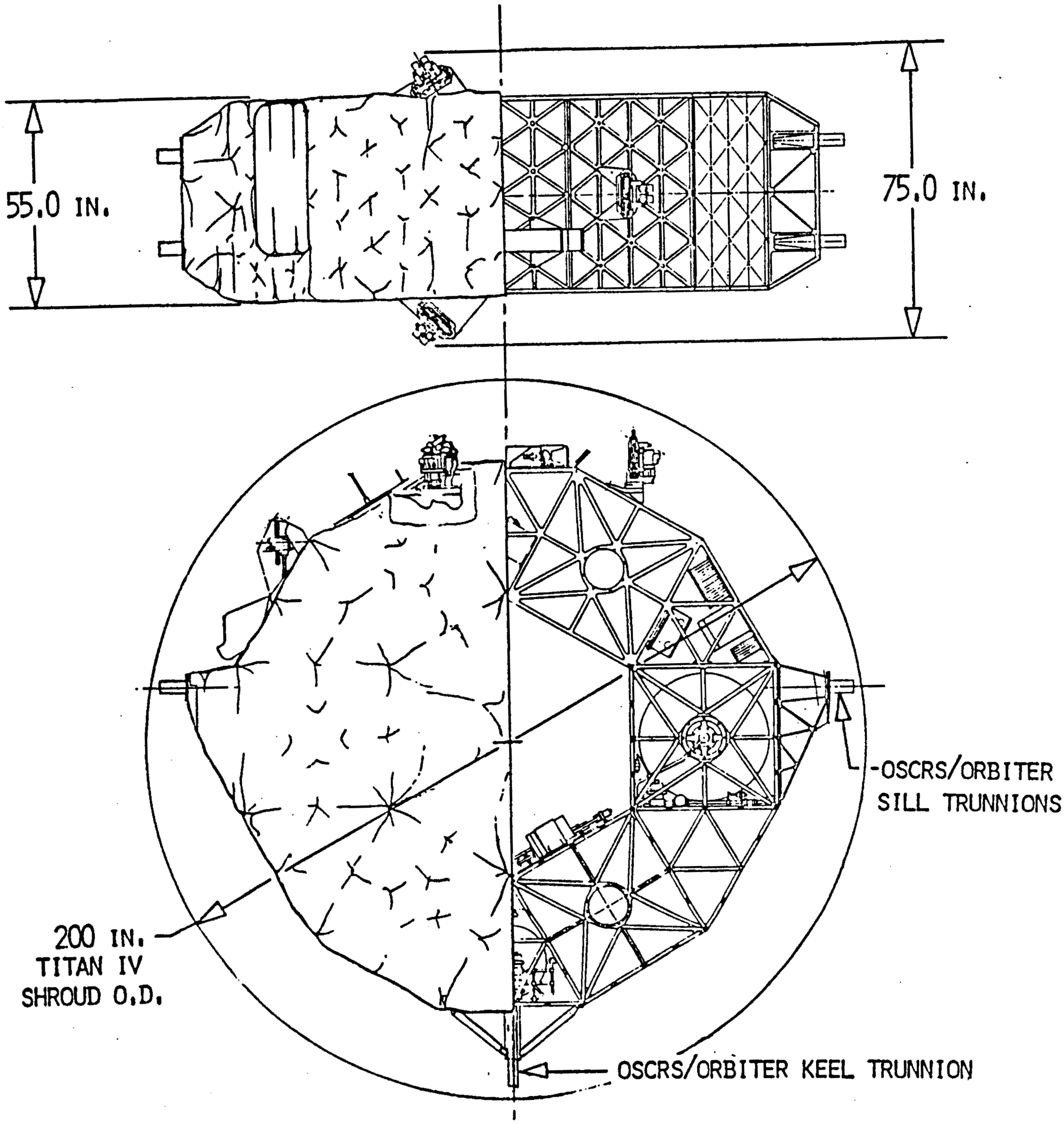
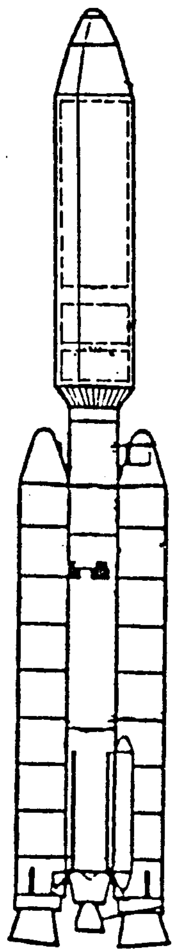


Figure 6.16-1 - Baseline OSCRS Dimensional Envelope



- CORE VEHICLE: 2 STAGE, 10FT DIA X 118 FT
- TWO PAYLOAD FAIRINGS: IUS; 200 IN. DIA X 56 FT.; CENTAUR; 200 IN. DIA X 86 FT.
- SHROUD: TRISECT DESIGN, ISOGRID CYLINDER, 7075-T351
  - JETTISONS IN FULL-LENGTH TRISECTORS SIMULTANEOUSLY
  - FIELD JOINTS IN CONSTANT SECTION PERMITS VARIABLE LENGTHS OF 20, 30, & 40 FT.
- ALLOWABLE LEO PAYLOAD LAUNCH CAPABILITY:
  - CENTAUR ———— 52,831 LBS.
  - CENTAUR PAYLOAD ———— 11,500 LBS.
  - PAYLOAD PECULIAR EQUIPMENT —— 313 LBS.

  $\boxed{\text{64,644 LBS.}}$
- PRESENT OSCRS / OMV WEIGHT:
  - OSCRS (BIPROP) ———— 11,900 LBS.
  - OMV ———— 12,722 LBS.

  $\boxed{\text{24,622 LBS.}}$ + ASE

Figure 6.16-2 - Selected OSCRS-ELV Launch Vehicle

SOLO-LAUNCH

Figure 6.16-3 illustrates three concepts for a solo OSCRS ELV launch. Concept A, in Figure 6.16-3, depicts an OSCRS-ELV interface utilizing pyrotechnic frangible nut fasteners at 6 to 12 places located at the existing structural points on the OSCRS outer perimeter.

In lieu of a 12 member open truss support structure interface at the ELV, a skin and stringer cone support is required with 36 bolt attachments to distribute the load into the ELV. This requirement applies to all payload to ELV interface structure.

The 3-piece payload shroud separates at this interface plane (payload fairing STA.00.00, launch vehicle STA. 165.00). Following shroud separation and launch vehicle main engine cut-off (MECO), the frangible nuts separate and the OSCRS is injected into a parking orbit by mechanical devices (i.e., springs, etc.). The TITAN-IV shroud, which is 200 inches in outside diameter, is required for all OSCRS launches. Modification to the shroud is required to provide clearance for the OSCRS-orbiter payload support trunnions. These trunnions permit OSCRS return to earth via the orbiter.

OSCRS SOLO-ORBIT

Launching an OSCRS solo (i.e., without an attached OMV or other spacecraft) may prove to be the simplest operational segment in placing and maintaining an OSCRS in a parking orbit. Once separated from the launch vehicle, the baseline OSCRS contains no provisions for attitude stabilization. There are many methods to provide active stabilization for orbiting spacecraft. All the methods employ unique subsystem characteristics. Candidate methods considered are:

1. Cold gas jet reaction control system (RCS).

2. Control moment gyros (CMG) which include:

   - o Reaction wheel assemblies
   - o Torquer wheels
   - o Gyroscopes

3. Magnetic moment devices (i.e., torque rods)

4. Gravity gradient/mass attraction

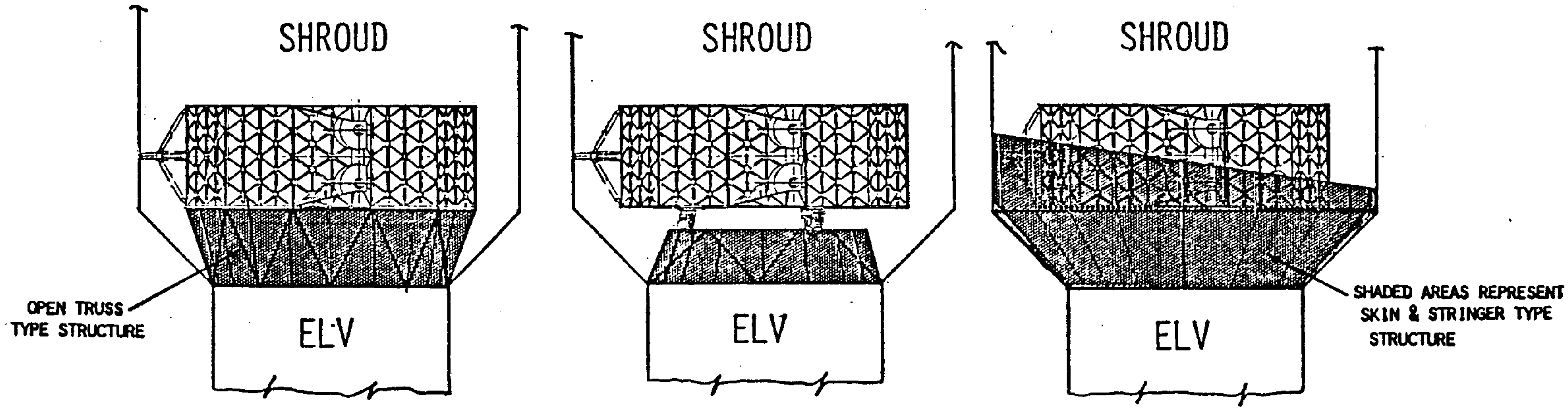
5. Spin-axis control

A detailed analysis by guidance, control and navigation specialists was beyond the scope of this study for the final selection of an attitude stabilization system.

(OSCRS SOLO LAUNCH)

MONO OR BIPROP OSCRS

MAXIMUM WEIGHT 11,900 LBS.



(A) PYRO SEPARATION

- HI-SHEAR FRANGABLE NUT FASTENERS (8) PLACES ON OSCRS.
- SKIN/STRINGER CONE SUPPORT OSCRS TO ELV.
- MINOR MODIFICATION TO BASELINE OSCRS STRUCTURE.

(B) FSS LATCHES

- MMF/FSS LATCHES (3) PERMANENTLY ATTACHED TO OSCRS STRUCTURE
- MINOR MODIFICATION TO BASE LINE OSCRS
- SKIN/STRINGER CONE SUPPORT TO ELV

(C) STS ACTION LATCHES

- LOADS CARRIED BY OSCRS SILL & KEEL TRUNNIONS
- NO OSCRS MODIFICATION
- SKIN/STRINGER CONE SUPPORT TO ELV

Figure 6.16-3 - ELV-OSCRS Interface Candidates

OSCRS/OMV COMBINATION LAUNCH

The alternative to the OSCRS solo launch is a launch combining the OSCRS and OMV. There are advantages to this simultaneous launch. First, all subsystems crossing the umbilical interface, primarily electronics and avionics, can be checked out on the ground prior to launch. Secondly, the OMV can provide the necessary attitude control for the OSCRS vehicle. The design, development and qualification testing of a separate OSCRS attitude control system is eliminated as well as the orbital rendezvous and berthing operations. On the other hand, depending on having an OMV available for every OSCRS mission does not make future operational sense since there will certainly be many more OSCRS launches than OMV launches. It would appear that launching an OSCRS, both solo and attached to the OMV however, is a valid requirement.

Two concepts were reviewed for a combination OSCRS OMV launch. One approach is to mate the OSCRS to the OMV utilizing one of the four existing payload attach points on the OMV, and take the TITAN booster launch loads through the OSCRS and OMV structure.

The OMV has provisions for (4) different payload attach points. Two are located on the forward face (i.e., translating end effector side) of the OMV. One interface contains (8) attach points on a 111.0 inch bolt circle and can support a cantilevered payload to a limit of 10,000 lb.-ft. The second interface provides (4) attach points on a 65.0 inch bolt circle and is limited to a 300 lb.-ft. limit load. Both sets of these attachments are accessible from the OMV aft face (i.e., solar panel face and location of the ΔV thrusters). However, the propulsion module (P/M) must be removed to expose both bolt circles.

Additionally, the OMV aft face contains (4) mounting points in the orientation of the (4) P/M module latches. The fourth payload attach provision is to utilize the P/M module latches themselves. Figure 6.16-4 illustrates these attach provisions.

Preliminary launch load calculations indicate that the two sets of bolt circles thru the OMV forward face (i.e., 111 and 65 inch diameters) can not support an OSCRS mated payload launch.

If the P/M module attach latches are utilized, requiring removal of the P/M, then the OMV vehicle ΔV thrust must come from the hydrazine cold gas 5-pound force thrusters since the four large 13-130 pound force thrusters are integral to the removed P/M module. This would greatly limit the OSCRS-OMV orbit transfer capability. A different approach utilizes the existing baseline OSCRS and OMV orbiter trunnion fittings to accept all launch loads while the OSCRS and OMV are mated through OMV's translating "see".

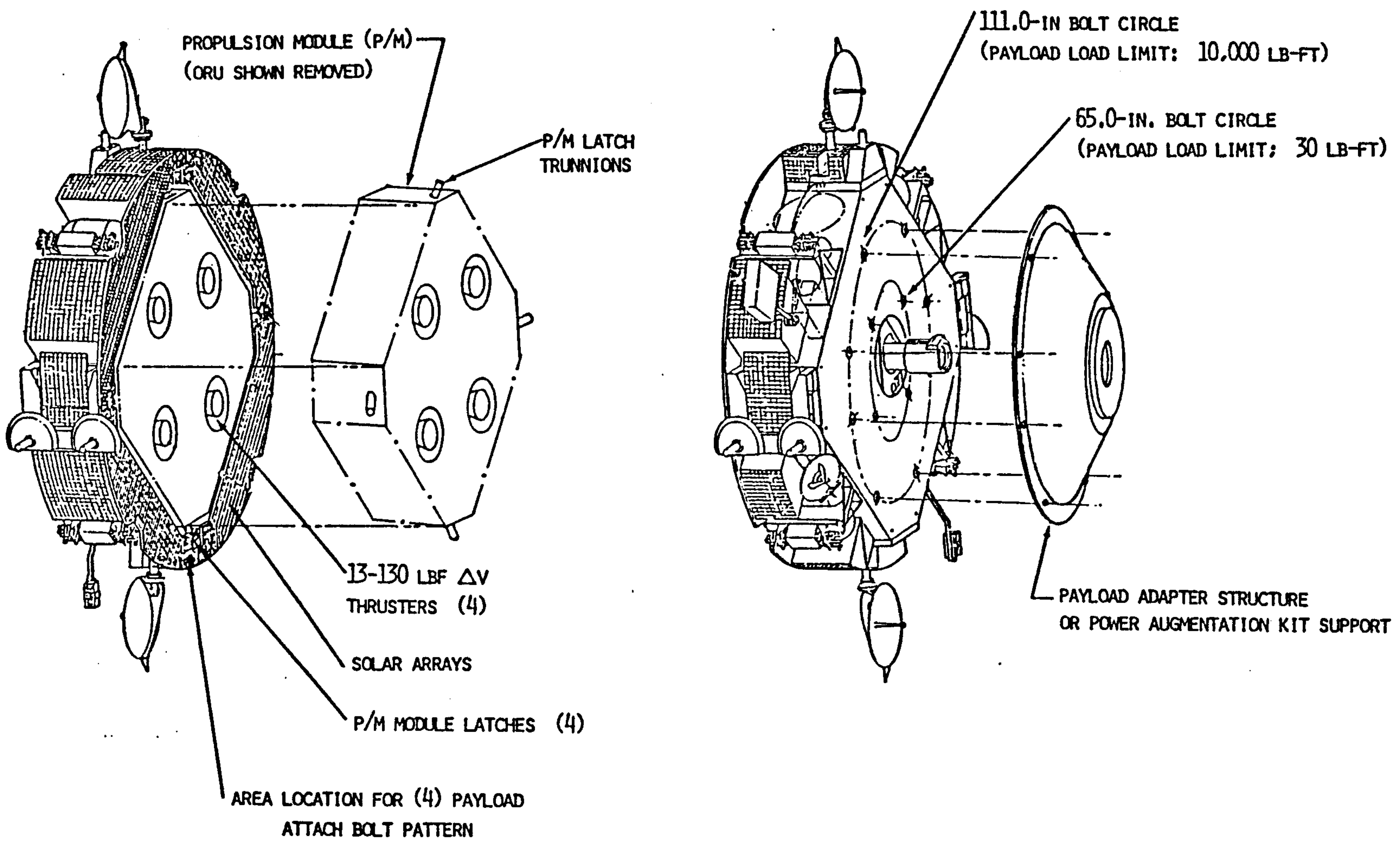


Figure 6.16-4 - OMV-Payload Attach Interfaces

The natural frequency (f_n) of a payload attaching to the upperstage of the TITAN IV, and within the 200 inch shroud, is approximately 4.5 H_Z. Both the OSCRS and OMV baseline f_n are dictated by the STS orbiter requirement of 6.3 H_Z, therefore, launch on the Titan IV should not present significant design/operational concerns.

As described earlier, the Titan IV shroud, 56 feet long, will more than accommodate an OSCRS/OMV vehicle. The aft separation of this shroud is at the Titan payload fairing station (PLF) 0.00 which is the interface to the TITAN second stage. This interstage shroud structure is a truncated cone (boat-tail) that is 52 inches long, 200 inches in outside diameter (O.D.) at the forward end and 120.53 inches O.D. at the Titan interface.

This boat-tail configuration necessitates a separate truncated cone support structure for OSCRS/OMV inside the existing shroud. In reality these are redundant structures and represent an unnecessary weight penalty. Figure 6.16-5 illustrates this redundant structure configuration.

Figure 6.16-6 presents a modified Titan-IV shroud configuration that allows the OSCRS/OMV launch support structure to double as the shroud base structure. This permits the shroud to separate at approximately PLF Sta. 231.00 and is in the 200 in. O.D. shroud constant section.

A similar shroud/payload integrated design is already in development by a Martin shroud customer. MDAC-HB presently provides to Martin, as the Titan payload adapter (TPA) program, the 200 inch shroud. This shroud is available in modular lenghts of 86, 76, 66 and 56 feet including both the bi-conic nose section, (19.9 feet long), and the boat-tail adapter/Titan interface (52 in. long).

The Martin customer obtains the 76 foot long shroud and removes the last 20 feet that includes the boat-tail/Titan interface. The customer then interfaces their payload support structure, including the Titan interface, with the new shortened shroud. The new payload support structure provides the point supports for the payload and doubles as the shroud at the same time. This is the same approach as shown in Figure 6.16-6.

ELECTRICAL POWER

There is no electrical power source on the baseline OSCRS. Any power requirement must eminate from a separate source and cross the OSCRS umbilical interface. OSCRS has power requirements for thermal control, avionics, data telemetry and, during consumables transfer, fluid/gas subsystem control.

SEPARATE ASE-SHROUD CONCEPT

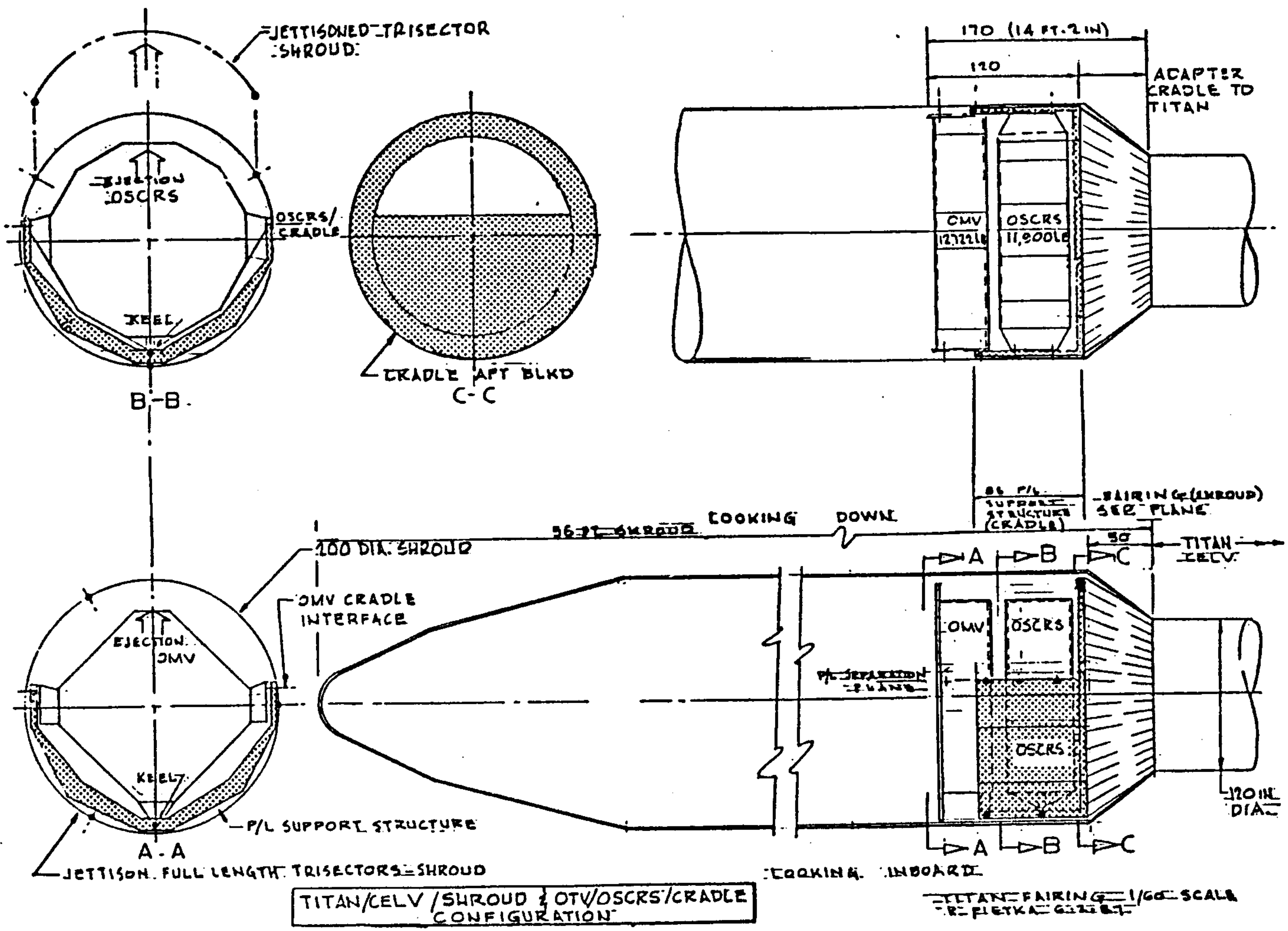


Figure 6.16-5 - ELV Launch Interface Configurations

INTERGRATED ASE-SHROUD CONCEPT

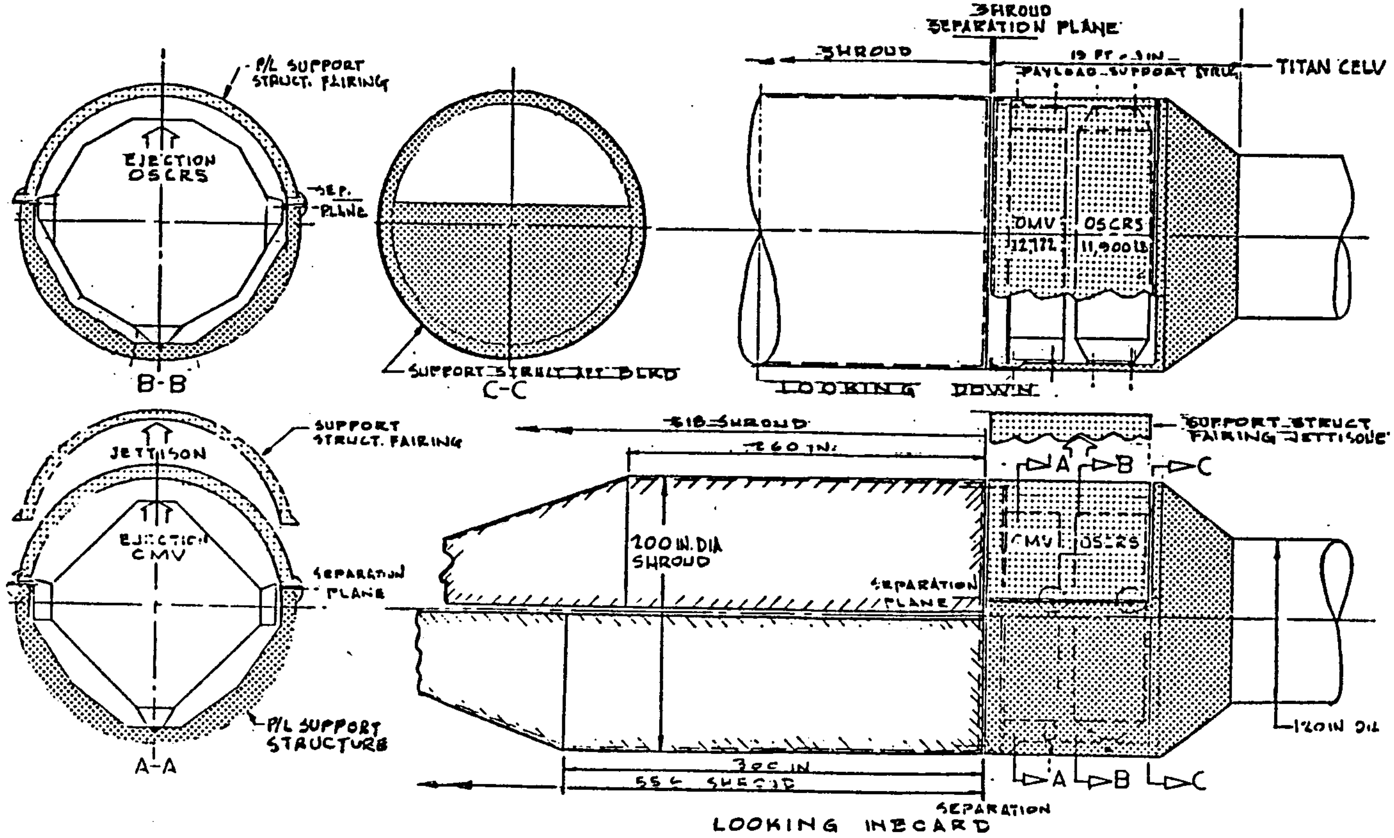


Figure 6.16-6 - ELV Launch Interface Configurations (Cont)

An OSCRS/OMV mission power analysis (section 6.6-1) showed a requirement for a maximum of 51 kwh for this 7-day mission ignoring the power requirements during the actual consumables transfer. The baseline OMV is able to supply 5 kwh of energy to OSCRS from its baseline power source (i.e., batteries). Therefore, the OMV can support about 16.5 hours of OSCRS on-orbit functions (i.e., thermal control and systems monitoring).

The need for an alternate method of providing electrical energy to OSCRS in support of long duration OSCRS/OMV missions is plain. There are several approaches.

TRW proposes a power augmentation kit concept. The kit, a cylindrical or truncated cone structure, can be mounted to either the front or aft face of the short range vehicle (SRV) segment of the OMV. Mounting a power kit on the OMV forward face is straight forward as the same 111 and 65 inch bolt circles are available and the same load limitations apply. The translating "see", which is in an orbital replacement unit (ORU) can be relocated forward of the forward face on an extending structure. Figure 6.16-4 illustrates this alternative.

A second approach is to add batteries to the OSCRS structure to supply the additional electrical requirements as shown in Figure 6.16-7. In addition, solar panels can also be added to the external perimeter if the battery requirement can be provided internal to the OSCRS structure.

The major conclusions and recommendations in this section are as follows.

Launching an OSCRS into a LEO parking orbit on an ELV, rather than in the STS orbiter bay, is a feasible concept. The ELV of choice is the TITAN IV vehicle and it is presently in production.

New mission unique payload support structure along with modification to the existing TPA shroud would be required. A similar modification is presently in progress for another non-related program utilizing the TITAN 200 in. x 76 ft. long shroud. The OSCRS modification would utilize the shorter 56 ft. long shroud.

Optimum utilization of the remaining empty forward shroud volume could be accomplished by the addition of other deployable payloads mounted forward of the OSCRS.

The OSCRS ELV launch scenarios should include solo, mated to the OMV, and mixed cargo manifesting launch capabilities. The payload-booster interfaces should utilize the STS orbiter sill and keel trunnions integral to both the OSCRS and OMV vehicle basic structure.

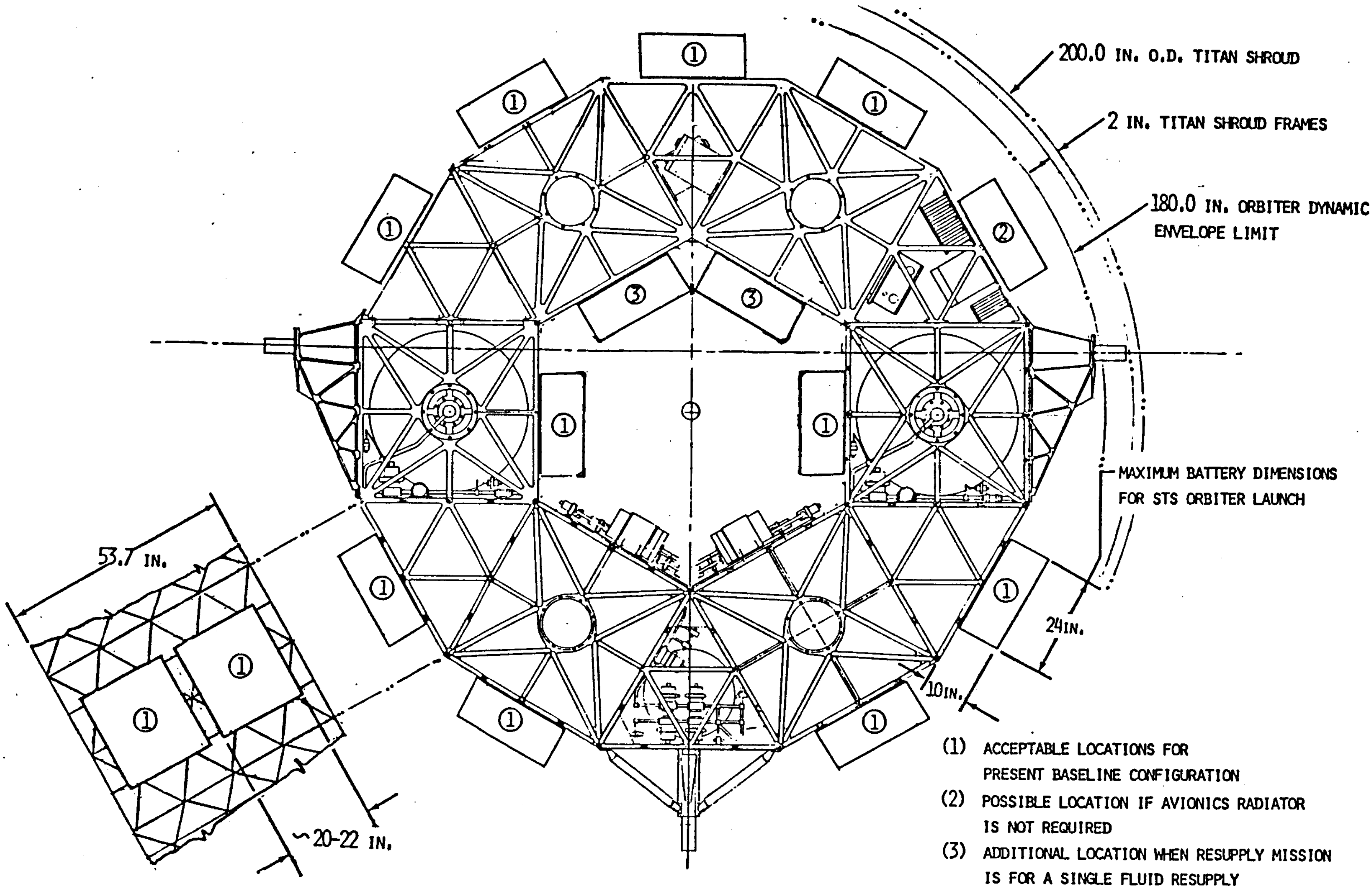


Figure 6.16-7 - Auxiliary Battery Power Supply Candidate Locations

Further evaluation of the developing shroud/payload combination support structure should determine if timely OSCRS/OMV inputs could result in utilization of this concept for OSCRS without major modifications to the shroud/payload structural concept. Major areas of interest would be the total payload capability, payload jettison schemes, and details of the payload support structure interface.

A careful review of the OMV's electrical capability to provide the power in support of OSCRS/OMV resupply missions is required.

6.17 Study Results Conclusions and Recommendations

6.17.1 Significant Conclusions

- Space Station resupply of hydrazine, MMH and NTO, and water should be provided by separate dedicated tankers. Station based tankers will resupply the Station, OMV and other spacecraft.

- Rockwell's OSCRS tanker concept is capable of resupplying presently defined fluid requirements.

- An OSCRS/OMV seven day mission will require an OSCRS power source.

- Present OSCRS thermal design will function at the Space Station.

- Offloading monopropellants and bipropellants (old tanker to replacement tanker at the Station) can be justified by both weight and cost considerations.

- Launching an OSCRS into a LEO parking orbit via an expendable launch vehicle (ELV) was determined to be a feasible concept and this will help relieve premium STS orbiter cargo manifesting.

- OSCRS design lends itself to remote operations. EVA activities should be limited to contingency operations.

- An industry standard umbilical interface was designed to combine all the interface functions for spacecraft capture, rigidization, retrieval, umbilical engagement and contamination control in one unified, sequentially controlled operation.

- A four string avionics system will provide automatic faultdown through two failures in a consistent manner and its ability to 'neck-up/neck-down' at a data interface makes the system more adaptable to usde with OMV and the Space Station.

- o It will take approximately $49 M and a 41 month lead time to design, develop, qualify, produce and deliver a dedicated Space Station based tanker with a water subsystem and a remote/automatic interface.

6.17.2 Recommendations

- o To develop at least two generic tankers to be based at the Space Station.
- o Positioning OSCRS at the Space Station should be planned to avoid extreme environmental conditions
- o Design OSCRS for both solo and OMV mated launches on expendable launch vehicles.
- o Select the Titan IV (ELV) which posses an abundant payload capacity and is in production.
- o Develop a standardized remote/automated umbilical interface.
- o A separate development program should be initiated for the standardized remote/automatic umbilical interface.
- o Plan $49 M and a 41 month leadtime to develop, qualify and deliver the second (water) Space Station based tanker.